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(NASA-CR-199586) MARS UPPER ATMOSPHERE DYNAMICS, ENERGETICS, AND EVOLUTION MISSION (MUADEE) TECHNICAL AND MANAGEMENT VOLUME. APPENDIX A: ATTACHMENTS Final Report (Lockheed Missiles and Space Co.) 299 p

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Lockheed Missiles & Space Company, Inc.
SUNNYVALE, CALIFORNIA

Mars Upper Atmosphere Dynamics, Energetics, and Evolution Mission

(MUADEE)

FINAL REPORT
TECHNICAL AND MANAGEMENT VOLUME

APPENDIX A: ATTACHMENTS

15 March 1994

MUADEE

TABLE OF CONTENTS

APPENDIX A ATTACHMENTS:

2/21/94 MUADEE Memorandum of Understanding Between The University of Michigan Space Physics Research Laboratory, the Jet Propulsion Laboratory, and Lockheed Missiles and Space Company

This memorandum of understanding (MOU) between the Mars Upper Atmosphere, Dynamics, Energetics & Evolution (MUADEE) mission Principal Investigator (PI)/University of Michigan, as chairman of the MUADEE Science Team, Jet Propulsion Laboratory (JPL), and the Lockheed Missiles and Space Company (Lockheed) establishes the basis for a cooperative effort to conduct the MUADEE project. The effective date of this Agreement is _____________.

MUADEE is a Discovery class mission to the planet Mars which will perform global orbital observations of the planet's upper atmosphere. The prospective mission calls for a single spacecraft to be launched and be placed in an orbit about Mars.

The MOU is mutually adopted by the parties to guide joint activities associated with the Discovery MUADEE project. While it is not a formal contractual agreement, it defines the key roles, responsibilities, relationships, and arrangements that underlie the management of the Discovery MUADEE project. These are as follows:

1. A Consortium has been formed between the PI, Lockheed, and JPL for the purpose of successfully proposing and carrying out the MUADEE mission, on schedule and within cost. The PI shall have overall responsibility, accountability, and decision authority and be responsible for the overall business, financial, technical and scientific management of the MUADEE project. An Oversight Board will consist of senior members from each consortium partner; the PI, as Chairman of the Oversight Board and leader of the Science Team, a senior representative of Lockheed, and a senior representative of JPL. The Oversight Board may appoint additional, non-voting members for specified purposes. The Oversight Board will operate by consensus.

2. The on-going work of the Oversight Board will be carried out by the Chairman. The University of Michigan will provide administrative support to the Oversight Board.

3. The Oversight Board will approve appointments of key project positions, including, but not limited to: the JPL Project Manager, the Lockheed Spacecraft Manager; and other key personnel assignees who will report directly to the PI. Key personnel will not be reassigned without concurrence from the PI.

4. The PI and Science Team are responsible for a detailed specification of scientific requirements, which are to be frozen at the time of the NASA Announcement of Opportunity for proposal submission. The PI and the Science Team will participate in any decisions that impact the mission science data return.

5. The PI and the Science Team are also responsible for the interpretation and distribution of the mission data and for publication of the results in the open literature. Proper credit will be given to NASA, JPL, Lockheed and other MUADEE University/Industry participants. The PI, JPL and Lockheed will not be restricted in their use of the mission data for public relations and educational purposes.

6. The PI is responsible for monitoring and fostering the necessary interactions between and among the team members: Lockheed, JPL and the Science Team. The PI, through the University of Michigan, will be responsible for the placement of the Spacecraft Contract with Lockheed.

7. JPL will assign a Project Manager (PM), approved by the PI. Although the Project Manager will report administratively to the JPL Office of Space Science and Instruments Discovery Office, he/she will be directly accountable and responsible to the PI; i.e., the PM will work for the PI.

8. The Project Manager's central role will be to ensure delivery of the instrumented MAUDEE spacecraft on schedule and within cost. The PM has primary responsibility over requirements at Level II and below. Requirements changes at Level I must be approved by the PI and will only be allowable to reduce cost and control schedule. Major technical issues are resolved through close collaboration of PI and PM. The PM will be responsible for producing concise, digestible project status summaries relating to cost, schedule and technical performance, science, spacecraft, instrument contractors, or university-provided hardware. The Project Manager will maintain an interface with the NASA Headquarters Discovery Program Office.

9. The University of Michigan will be responsible for mission operations. Navigation, tracking and data acquisition will be centered at JPL.

10. Lockheed will be responsible for MUADEE system integration, spacecraft development, integration, testing, and launch operations. Lockheed will appoint a Spacecraft Manager, subject to the approval of the Oversight Board, with the responsibility for the entire Lockheed effort.

11. The following assumptions apply to the MUADEE Proposal Development:

a. The PI, with the assistance of a Proposal Team Leader, is responsible for the preparation of the joint PI/JPL/Lockheed MUADEE Discovery Proposal in response to the Discovery Announcement of Opportunity (AO).

b. The PI, the Science Team Members, JPL and Lockheed will supply the necessary engineering, management,. technical, and other services as well as nonproprietary cost information, exhibits, designs, and plans related to the work they propose to perform at their own expense in support of the MUADEE Proposal. Proprietary data will be handled by Individual Proprietary Information Nondisclosure Agreements between Lockheed, JPL, the PI and appropriate Science Team Members to ensure that such proprietary data is controlled within each of the consortium members facilities.

c. The final proposal will be reviewed by the Oversight Board, and the Individual Oversight Board Members will obtain internal concurrences, if required, prior to the submission of the MUADEE Proposal to NASA. All contacts with NASA pertaining to the preparation of the proposal will be made through the PI.

12. The following assumptions apply to program costs:

a. The parties to this Memorandum recognize the special requirements imposed by the cost constraints on Discovery missions. The MUADEE Oversight Board will exercise oversight over all aspects of MUADEE expenditures.

b. The type of contact to be used for the Lockheed Spacecraft effort will be a cost plus award vehicle consistent with the MUADEE program cost and the overall Discovery philosphy. The contract will incorporate on-orbit performance incentives and descope options to control costs to a not-to-exceed ceiling. Under such a contract with Lockheed the ceiling will be established consistent with the overall MUADEE cost assumptions and as agreed to by the Oversight Board.

c. Product Assurance for the flight elements will be provided by Lockheed according to established Lockheed practices and procedures, with technical oversight by the MUADEE Project Manager. The Product Assurance Plan is subject to approval by the JPL MUADEE Project Manager prior to submission of the final program cost proposal.

13. Nothing in this agreement shall be deemed to constitute, create, give effect to or otherwise recognize a joint venture, or formal business entity of any kind, and the rights of the parties hereto

shall be limited to those expressly set forth herein. Nothing herein shall be construed as providing for the sharing of profits or losses arising out of the efforts of the consortium partners, except as may be provided for in any contract agreed to by and between the PI, JPL and Lockheed.

14. Any news release, public announcement, advertisement or publicity proposed to be released by any of the parties to this agreement concerning the activities of the other party in connection with the MUADEE Proposal or any resulting contract shall be subject to the approval of the other party(s) prior to release.

15. This agreement constitutes the entire understanding and agreement of and between the consortium with respect to the MUADEE Mission and any or all changes hereto require that such changes be in writing and be unanimously agreed to by the Oversight Board prior to any implementation by any of the parties.

IN WITNESS WHEREOF, the parties hereto have caused this Agreement to be executed as of the day and year first above written.

____

Principal Investigator (PI)
Dr. T. L. Killeen
University of Michigan
Ann Arbor, Michigan 48109-2143

____

Jet Propulsion Laboratory
California Institute of Technology
Pasadena, California 91109-8099

____

Lockheed Missile and Space Company
Sunnyvale, California

18 February 94

MARS UPPER ATMOSPHERE DYNAMICS, ENERGETICS AND EVOLUTION MISSION

STATEMENT OF WORK

1.0 Introduction

This Statement of Work (SOW) defines the spacecraft contractor's efforts to implement the MUADEE spacecraft. The main purpose of this program is to:

- Design and develop the spacecraft bus
- Integrate and test the spacecraft
- Deliver said craft to the Eastern Range
- Provide launch and operations support

The spacecraft contractor shall furnish the necessary personnel, facilities, services and materials required to accomplish the foregoing tasks.

In accomplishing the development of the MUADEE spacecraft, the contractor shall:

- Provide the spacecraft bus
- Receive the instruments as supplied by the University of Michigan Space Physics Research Laboratory and integrate them with the bus to form the spacecraft and perform spacecraft level testing
- Provide all necessary ground support equipment
- Provide a spacecraft electrical interface simulator for the instruments to checkout the electrical interface at the instrumenter's facility prior to delivery
- Support the combined testing of the spacecraft, ground system and the DSN (end-to-end testing) before launch.
- Deliver the spacecraft to the Eastern Launch site
- Provide launch and post-launch support
- Support flight operations planning, including defining flight operations requirements
- Provide spacecraft operator training and supervision for 30 days after launch
- Provide the spacecraft bus engineering data base required for flight operations
- Support on-orbit checkout and initial calibrations of the instrument and spacecraft bus
- Perform the necessary systems engineering in support of all of these tasks

2.0 Specific Areas of Effort

2.1 Mission Planning Support

Spacecraft contractor will assist, where appropriate and possible, with mission planning. Inasmuch as mission goals are affected by spacecraft design, the spacecraft contractor shall maintain a close supporting role with the University of Michigan through regular teleconference and videoconference contact, reports and periodic reviews. The full implications of design specifications (and design changes) shall be communicated to the principal investigator (U. Mich.) as quickly as possible (see Section 2.8.2 on Configuration Management).

2.2. Spacecraft Subsystem Design

The spacecraft contractor shall define a baseline for the spacecraft bus to be used on the mission. This shall form the basis for the mission, understanding that some cost effective modifications may be proposed in order to accommodate changes in instrument requirements. Specifications for the spacecraft are given in Document (   ). This is the "spacecraft definition period" and will commence on the contract award date or initiation of the authority to proceed.

The spacecraft contractor shall perform all engineering tasks needed to define the design including all analyses and system trades. Specific subsystems which should be addressed are:

2.2.1 Structures and Mechanisms

Spacecraft contractor shall provide all necessary engineering analysis and trades required to develop suitable structures and mechanisms for the spacecraft bus including all on-orbit deployable mechanisms which may be used.

2.2.2 Thermal Considerations

The spacecraft contractor shall provide engineering support as required, including analysis and trades, to develop a thermal design which is suitably accommodated to the instruments, posing no threat of thermal distortion, either mechanical or electronic.

2.2.3 Command and Data Handling

The spacecraft contractor shall provide all necessary engineering support to develop a design for the command and data handling subsystem.

2.2.4 Guidance, Navigation and Control

The spacecraft contractor shall provide all necessary engineering support to develop a design for the guidance, navigation and control subsystem.

2.2.5 Propulsion Subsystem

The spacecraft contractor shall provide all necessary engineering support to develop a design for the propulsion subsystem. Contractor shall include a detailed analysis of the "delta v" requirements for the MUADEE spacecraft and the corresponding propulsion budget dictated thereby.

2.2.6 Electrical Power and Distribution

The spacecraft contractor shall provide all necessary engineering support required to fabricate, assemble and test the electrical power subsystem and its distribution subsystem.

2.2.7 Communications

The spacecraft contractor shall provide all necessary engineering support required to fabricate, assemble and test the communications subsystem for the MUADEE spacecraft.

2.3 Accommodation of Instruments, Mechanisms and Subsystems

2.3.1 Spacecraft contractor shall exercise suitable care to see that instruments and subsystems are accommodated properly. Areas which should be considered include:

- Structures and Mechanisms
  - Weight, mounting locations
  - Disturbance management
  - Field-of-view blockage
- Power - availability and distribution
  - Power interface, budgets, instrument power sharing
- Thermal
  - Heat transfer and "smoothing"
  - Radiator field-of-view blockage
  - Special surface treatment
- Contamination
  - Earth contamination/cleanliness
  - Optical surfaces
  - Venting, plume contamination
  - Planetary protection
- Test Requirements
  - Planning and facilitating tests, data and postprocessing
- Command and Data Handling
  - Data handling and storage
  - Processing rates
  - Radiation shielding
  - Critical systems management

2.4 Integration and Test

The spacecraft contractor shall furnish all necessary personnel, facilities, equipment, services and materials required to support integration and test efforts for the MUADEE spacecraft.

This includes all electrical, mechanical and thermal integration. Contractor shall also provide necessary technical efforts to assure that problems are adequately addressed as they arise.

Facilities may include office space and support, clean rooms, and space suitable for bench acceptance tests. The spacecraft contractor shall mount, align and verify that subsystems and instrumentation are accurately aligned.

2.4.1 Functional Tests

Contract shall provide all resources necessary to define, document and analyze all functional and performance tests. Contractor shall define each test and prepare and operate the facility.

2.4.2 Environmental Tests and Contamination Control

Spacecraft contractor shall provide all resources necessary to define, document and analyze the needed environmental tests. Contractor shall define each test and prepare and operate the facility. Such tests are to be accommodated with a suitable testing period. Contractor shall also be responsible for monitoring contamination (both particulate and molecular) of the spacecraft from the beginning of integration through the launch.

2.4.3 Ground Support Equipment (GSE)

Spacecraft contractor shall furnish both mechanical and electrical GSE (or provide all resources necessary to design and fabricate such equipment).

2.4.4 Handling and Transportation

Contractor shall provide all resources necessary to transport the spacecraft and all necessary equipment to the launch site. This shall be done safely and in a manner compatible with applicable environmental specifications. Contractor shall also provide similar transportation and handling back to a suitable facility after the launch.

2.4.5 Eastern Range Operations

Spacecraft contractor shall provide all resources necessary to analyze, plan, perform, coordinate and document all phases of launch operations for the MUADEE spacecraft from arrival at the Range through the launch, including spacecraft servicing in the event of mission abort. Contractor responsibility extends 30 days beyond the launch date.

2.5 Instrument and Launch Vehicle Interface Coordination

2.5.1 Instrument Interface Accommodation

The spacecraft contractor shall perform all systems analysis and engineering tasks required to define all aspects of the spacecraft bus-to-instrument interface and the spacecraft-to-launch-vehicle interface. Contractor shall negotiate Interface Control Documents (ICD's) with the MUADEE scientific instrument manager using the appropriate Instrument Definition Document (IDD) as a basis. The negotiated ICD shall ultimately replace the appropriate IDD.

2.5.2 Launch Vehicle Interface

The spacecraft contractor shall exercise due diligence in the construction of the interface between the spacecraft and the launch vehicle. This includes all engineering support as well as the equipment needed to fabricate, assemble and test an interface simulator which is to be delivered to the University of Michigan facility for checkout of the launch vehicle interface.

2.6 Mission Operations

The spacecraft contractor shall provide assistance to the University of Michigan personnel, where possible.

2.6.1 Training

During the development of the spacecraft, contractor shall provide training for University of Michigan personnel in all phases necessary for mission operations. This shall include preparation of training materials for spacecraft operations, instruments and, in the checkout phase, the spacecraft interface simulator.

2.6.2 Post-Launch Support

Contractor shall provide training and operations support for a period of thirty (30) days after launch.

2.6.3 On-Call Support

Phase E of the MUADEE project is the period after the 30-day post-launch phase. During this time, the spacecraft contractor will be available for consultation and assistance in operations.

2.7 Launch Operations

The spacecraft contractor shall be responsible for directing flight operations from launch through orbital checkout. The spacecraft contractor shall perform this task by advising the University of Michigan Flight Operations Team (who will be operating the MUADEE console).

2.8 Management Planning

The spacecraft contractor shall perform the necessary direct management functions and provide a management structure responsible for overall project control to assure that all requirements of this Statement of Work are accomplished successfully and in a timely manner. A full-time Program Manager shall be appointed who shall have sufficient corporate authority to assure that contract cost, schedule and technical requirements are fully met.

2.8.1 Program Reviews

Periodic reviews shall be held to communicate the status of the program and facilitate recommendations and changes (if needed). Additional reviews and meetings shall be scheduled as needed. The location of program reviews and meetings shall be determined by mutual consultation and approval.

2.8.2 Configuration Management

Spacecraft contractor shall establish, implement and maintain a configuration management system to ensure that all applicable changes are reviewed in a systematic manner to determine their impact on performance, schedule and cost.

2.8.3 Project Schedules

The spacecraft contractor shall establish, implement and maintain a resource management system for planning authorizing and controlling all the resources of the MUADEE program. Such work will be documented on schedules which will provide immediate visibility into manpower, cost and performance.

3/4/94

MISSION OPERATION DOCUMENT

FOR

THE MUADEE SPACECRAFT PROGRAM

Prepared by : A. Binder

TABLE OF CONTENTS

1.0 Introduction

Section 2.0 contains Version 1.0 of the MAUDEE Mission Operation Document (MOD). The MOD contains the currently defined mission rules, operational procedures and commands needed to control the spacecraft, its subsystem and the science payload. As the subsystems, science instruments and science mission definitions mature, and as construction, testing and training proceed, the MOD will continually be expanded and be upgraded.

MAUDEE
Mission Operations Document

Version 1.0

Alan Binder
Mission Operations Manager
Feb. 4, 1994

1. General Mission Rules

1.1. The Spacecraft will always be kept in a fail-safe condition.

1.1.1. The Science Instruments will be turned to standby or, if necessary, off during periods of non-nominal Spacecraft conditions.

1.1.2. The Spacecraft Communication Subsystem will be switched to the 8 bps, engineering data rate downlink mode using the omni antenna during non-nominal Spacecraft conditions.

1.1.3. Any Science Instrument which fails to function properly will be turned off for the duration of the mission or until the failure can be corrected.

1.1.4. Any Spacecraft Subsystem, for which there is a backup, which fails to function properly will be turned off for the duration of the mission or until the failure can be corrected.

1.1.5. Mars orbital maneuvers will be conducted in a manor to minimize the possibility that the Spacecraft to enter the Martian atmosphere.

1.1.6. Electrical power usage will never be such that the battery depth of discharge is >60%.

1.2. Uplink Commands

1.2.1. **The Mission Director shall possess override authority for all commands in all instances.**

1.2.2. **All commands issued to the Spacecraft will be issued through the Command Officer, after authorization by the on-duty Flight Director.**

1.2.3. **All commands must be verified by the Verification Team before being uplinked to the Spacecraft.**

1.2.4. **Nominal Uplink Commands**

1.2.4.1. Uplink activities will occur within view of a tracking station with command capability.

1.2.4.2. No commands, except fail-safe commands, will be uplinked during the 10 minute plus light time S/C check-out period following S/C acquisition after loss of signal, due to occultation or any other event.

1.2.4.3. Command Sequence

- Each uplink command or command series will be verified by the S/C before the command or command series is accepted by the S/C.
- Immediately after S/C acceptance of a command or command series, the command or command series will be transmitted to Earth for verification by the Command Officer and the Verification Team.
- If the command or command series found to be incorrect by the Command Officer and the Verification Team, the command uplink procedure will be repeated until the command or command series is properly accepted by the S/C.
- When the command or command series is correct, it will be carried out or stored for later execution.
- There will be a 10 minute plus light time S/C check-out period after each command or command series that is uplinked. No further commands or command series can be uplinked or executed during these 10 minutes plus light time.
- Uplink of a command or command series may not occur if a loss of signal is expected, due to the occultation of the S/C or any other event, before the end of the 10 minute plus light time verification period.

2. Nominal Mission Operations

2.1. Launch Through Spacecraft Separation/Turn On

2.1.1. During launch the Spacecraft will be in a low or no power, survival mode in which the Spacecraft neither transmits data nor receive commands.

2.1.2. Prior to third stage ignition, the Spacecraft and the third stage will be spun up to TBD rpm by a launch vehicle subsystem.

2.1.3. At separation from its launch booster, the Spacecraft and all its subsystems, except the Science Instruments, will be turned on. Using the omni antenna, the Spacecraft will transmit engineering data at the 10 kbps data rate and receive commands.

2.2. Trans-Mars Coast

2.2.1. **The 20 minute period immediately after separation will be used to determine the condition of the Spacecraft and all of its engineering subsystems.**

2.2.2. **20 minutes after separation, the high gain antenna will be deployed.**

2.2.3. **The Spacecraft will be spun down to 3 rpm 10 minutes after antenna deployment.**

2.2.4. **10 minutes after spin down, the Spacecraft will be reoriented from its launch orientation to its cruise attitude with +Z in the north direction of the Mars transfer orbit.**

2.2.5. **Science Instruments**

2.2.5.1. The Science Instrument initial turn on sequence will begin 10 minutes after the Spacecraft is in its cruise attitude (see 3.).

2.2.5.2. The Science Instruments will be turned on once per month during the cruise (see 3.).

2.2.6. **There will be 3 trajectory correction maneuvers.**

2.2.5.1. The first Trajectory Correction Maneuver will occur 2 week after launch.

2.2.5.2. The second Trajectory Correction Maneuver will occur 8 weeks before Mars orbit injection.

2.2.5.3. The third Trajectory Correction Maneuver will occur 1 month before Mars orbit injection.

2.3. Mars Orbit Insertion Maneuver

2.3.1. The MOI maneuver sequence will consist of one burn centered on the periapsis passage of the incoming hyperbolic orbit.

2.3.2. The MOI Burn

2.3.2.1. The MOI burn will put the S/C into its initial elliptical capture orbit.

- 250 km altitude periapsis.
- 33,100 km altitude apoapsis.
- 99.5 deg. inclination.
- 1 day period.

2.3.2.2. S/C preparations, the uplink, storage and verification of the burn commands for the MOI maneuver must all be completed ***no less than 3 days*** before loss of signal as the S/C is occulted by the Mars during its approach to its first periapsis passage or before the burn begins, which ever comes first.

2.3.6. MOI burn duration must be less than 35 minutes.

2.4. Mars Orbit

2.4.1. Initial Orbital Operations

2.4.1.1. The S/C will remain in its MOI burn/aerobraking attitude during this entire phase.

2.4.1.2. Preparations for aerobraking will begin 2 days after MOI:

- Periapsis altitude will be lowered in three or more steps down to the altitude required to reduce the apoapsis altitude to 2500 km in 150 days or less.
- The periapsis adjustment burns will occur no more frequently than once every two orbits.
- Science Instruments will be on stand or OFF by during all burn sequences.

2.4.1.3. Science data will be collected every orbit (see 3.).

2.4.1.4. The spin rate of the S/C will be 3 rpm.

2.4.2. First Aerobraking Phase (approximately 145 days)

2.4.2.1. The S/C will be in its aerobraking attitude during this entire phase.

2.4.2.2. Periapsis altitude adjustment burns will be made as required to keep the apoapsis decay rate at the desired value.

2.4.2.3. Aerobraking will end when the apoapsis altitude is 2500 km.

2.4.2.4. Periapsis altitude will be increased to approximately 130 km when the 2500 km apoapsis altitude is reached.

2.4.2.5. Science data will be collected every orbit (see 3.).

2.4.2.6. The spin rate of the S/C will be 3 rpm.

2.4.3. Latitude Survey Phase (approximately 480 days)

2.4.3.1. The S/C will be in its cartwheel attitude during this entire phase.

- The spin axis of the S/C will be 90+/- 5 deg. to the plane of its orbit.
- S/C attitude adjustment maneuvers will be made as required to keep the 90+/- deg. cartwheel attitude as the orbit percesses.

2.4.3.2. Orbit adjustment burns will be made as required to keep the apoapsis at 2500.

2.4.3.3. S/C Spin Rates

- The S/C will have a 3 rpm spin rate when science data are not acquired.
- The S/C will have a 1 rp-orbit spin rate when science data are acquired.

2.4.3.4. Science data will be collected every 4 orbits during this phase (see 3.).

2.4.3.5. Solar Conjunction

- There will be a period around solar conjunction (approximately 40 days) near the middle of this phase when communications with the S/C are not possible..
- The S/C will be put in its hibernation mode during the solar conjunction period.
- If required, the periapsis altitude will be raised to a "safe altitude" during the solar conjunction period.

2.4.4. Second Aerobraking Phase (approximately 30 days)

2.4.4.1. The S/C will be in its aerobraking attitude during this entire phase.

2.4.4.2. Periapsis Altitude

- Periapsis altitude will be lowered in one or more steps down to the altitude required to reduce the apoapsis altitude to 200 km in 30 days or less.

- Periapsis altitude adjustment burns will be made as required to keep the apoapsis decay rate at the desired value.
- Periapsis altitude will be increased to 200 km when the 20 km apoapsis altitude is reached..

2.4.4.3. Science data will be collected during every 4 orbits (see 3.).

2.4.4.4. The spin rate of the S/C will be 3 rpm.

2.4.5. Diurnal Phase (approximately 120 days)

2.4.5.1. The S/C will be in its cartwheel attitude during this entire phase.

- The spin axis of the S/C will be 90+/- 5 deg. to the plane of its orbit.
- S/C attitude adjustment maneuvers will be made as required to keep the 90+/- deg. cartwheel attitude as the orbit percesses.

2.4.5.2. Orbit adjustment burns will be made as required to keep the orbit circular at 200.

2.4.5.3. S/C Spin Rates

- The S/C will have a 3 rpm spin rate when science data are not acquired.
- The S/C will have a 1 rp-orbit spin rate when science data are acquired.

2.4.5.4. Science data will be collected every 4 orbits during this phase (see 3.).

2.4.6. End of Mission - Quarantine Orbit

2.4.6.1. After the end of the Diurnal Phase, the Spacecraft will be put into a 250 x 1000 km altitude quarantine orbit.

2.4.6.2. The S/C will be oriented with its spin axis perpendicular to the Martian heliocentric orbit.

2.4.6.3. The S/C will be put into its hibernation mode.

3. Nominal Science Operations

3.1. Science Mission

3.1.1. Science Suspension

3.1.1.1. Suspension of nominal science activities shall be minimized.

3.1.1.2. Science activities will be suspended during the operational phases of:

- Attitude maneuvers.
- Trajectory correction maneuvers.
- Mars orbit insertion maneuvers.
- Mars orbit periapsis and apoapsis correction maneuvers.
- Spin up/down maneuvers.

3.1.1.3. Science activities will be suspended when required by power and downlink constraints.

3.1.1.4. Science activities will be suspended during Solar Conjunction blackout.

3.1.1.5. Science activities will be suspended at the End of Mission, when the S/C is put into its quarantine orbit.

3.1.2. Nominal Science Data Acquisition

3.1.2.1. Trans-Mars Cruise Phase

- The VMAG will collect data for 7 days every 4 weeks (see 3.).
- The EUV will collect data for once per day (see 3.).
- No other Science Instruments will collect Science Data .

3.1.2.2. Initial Orbital Phase

- The FPI and UVS sensors will collect data every orbit when the S/C is at altitudes <1000 km around periapsis passage.
- The UVS sensors will collect full disk scan imaging data of Mars every orbit when the S/C is near apoapsis passage.
- The IP sensor will collect data every orbit when the S/C is both at altitudes <1000 km around periapsis passage and on the day side of Mars.
- The NMS, RPA/IDA and LP/EUV sensors will collect data every orbit when the S/C is within +/-30 minutes of periapsis passage.
- The VMAG will collected data during the entire orbit (when allowed by power and downlink constraints).

3.1.2.3. First Aerobraking Phase

- The FPI and UVS sensors will collect data every orbit when the S/C is at altitudes <1000 km around periapsis passage.
- The UVS sensors will collect full disk scan imaging data of Mars every orbit when the S/C is near apoapsis passage.
- The IP sensor will collect data every orbit when the S/C is both at altitudes <1000 km around periapsis passage and on the day side of Mars.
- The NMS, RPA/IDA and LP/EUV sensors will collect data every orbit when the S/C is within +/-30 minutes of periapsis passage.
- The VMAG will collected data during the entire orbit (when allowed by power and downlink constraints).

3.1.2.4. Latitude Survey Phase

- Science data will be acquired on each orbit allowed by the power and downlink constraints, but never less frequent than once every 4 orbits.
- The FPI and UVS sensors will collect data when the S/C is at altitudes <1000 km around periapsis passage.
- The IP sensor will collect data when the S/C is both at altitudes <1000 km around periapsis passage and on the day side of Mars.
- The NMS, RPA/IDA, LP/EUV and VMAG sensors will collect data when the S/C is within +/-30 minutes of periapsis passage.

3.1.2.5. Second Aerobraking Phase

- Science data will be acquired on each orbit allowed by the power and downlink constraints, but never less frequent than once every 4 orbits.
- The FPI and UVS sensors will collect data when the S/C is at altitudes less <1000 km around periapsis passage.
- The IP sensor will collect data when the S/C is both at altitudes less <1000 km around periapsis passage and on the day side of Mars.
- The NMS, RPA/IDA, LP/EUV and VMAG sensors will collect data when the S/C is within +/-30 minutes of periapsis passage or at altitudes <1000 km.

3.1.2.6. Diurnal Survey Phase

- Science data will be acquired on each orbit allowed by the power and downlink constraints, but never less frequent than once every 4 orbits.
- The FPI, UVS, NMS, RPA/IDA, LP/EUV and VMAG sensors will collect data continuously during the allowable orbit(s).
- The IP sensor will collect data when the S/C is on the day side of Mars during the allowable orbit(s).

3.2. Science Instrument Sequences

3.2.1. Initial Turn On Sequence

3.2.1.1. Plasma Instrument Package

- The PIP will be switched ON within 2 hours after TMI for an initial 1 day instrument checkout.
- The VMAG will collect 7 days of data in the geomagnetic field and interplanetary field 10 minutes after the PIP is switched ON.
- The EUVs will start collecting data once per day 10 minutes after the VMAG is switched ON.
- The LPs be deployed and switched ON for an initial 1 day checkout 10 minutes after the NMS cap is broken off (Note: after MOI, see 3.2.1.2.).

3.2.1.2. Neutral Atmosphere Package

- The NAP will be switched ON within 2 hours after MOI for an initial 1 day instrument checkout.
- The FPI will be turned ON for check out 10 minutes after the NAP is switched ON.
- The covers of the FPI telescopes will be deployed 10 minutes after the FPI is switched ON.
- The NMS will be turned ON for checkout 10 minutes after the covers of the FPI telescopes are deployed.
- The NMS cap will be broken off 10 minutes after the covers of the NMS is switched ON.

3.2.1.3. Scanning Imaging Package

- The SIP will be switched ON 1 day after TMI for an initial 1 day instrument checkout.
- The IP will be turned ON for check out 10 minutes after the SIP is switched ON
- The cover of the IP will be deployed 10 minutes after the IP is switched ON.
- The UVS will be switched ON for checkout 10 minutes after the IP cover is deployed.

3.2.2. Nominal Trans-Mars Cruise Operations

3.2.2.1. Plasma Instrument Package

- The VMAG will collect data for 7 days every 4 weeks.
- TheRPA/IDM will have a health chech once every 4 weeks.
- The EUV will collect data once per day.

3.2.2.2. Neutral Atmosphere Package will be OFF during cruise.

3.2.2.3. Scanning Imaging Package

- The NMS will collect calibration data for 8 hours once every 4 weeks.
- The LP will collect calibration data once every 4 weeks

3.2.3. Nominal Mars Orbit Operational Sequences

3.2.3.1. Aerobraking Mode:

- NAP, TBD.
- SIP, TBD.
- PIP, TBD.

3.2.3.2. Cartwheel Spin Mode:

- The FPI and UVS sensors will collect data when the S/C is at altitudes <1000 km around periapsis passage.
- The IP sensor will collect data when the S/C is both at altitudes <1000 km around periapsis passage and on the day side of Mars.
- The NMS, RPA/IDA, LP/EUV and VMAG sensors will collect data when the S/C is within +/-30 minutes of periapsis passage.

3.2.4. Magnetometer Boom Deployment Sequence TBD

3.3. Science Instrument Operations

3.3.1. **All Science Instruments operational commands, not related directly to Spacecraft/Mission operations, will be initiated within the on-duty Command and Control Team by the Science Flight Controller at the request of the Science Teams.**

3.3.2. Neutral Atmosphere Package Commands

3.3.2.1. NAP
- ON.
- OFF.

3.3.2.2. FPI
- ON.
- OFF.
- Deploy Telescopes' Covers.
- Calibrate.

3.3.2.3. NMS
- ON.
- OFF.
- Break off cap.
- Neutral ON.
- Neutral OFF.
- Ion.

3.3.3. Scanning Imaging Package Commands

3.3.3.1. SIP
- ON.
- OFF.

3.3.3.2. UVS
- ON.
- OFF.
- Spectral Mode.
- Wavelength Mode.
- Stare Mode.
- Imaging Mode.

3.3.3.3. IP
- ON.
- OFF.
- Deploy Cover.
- Wavelength.

3.3.4. Plasma Instrument Package Commands

3.3.4.1. PIP
- ON.
- OFF.

3.3.4.2. RPA/IDM
- ON.
- OFF.
- Mass Search.
- RPA Scan.
- Analyzer Scan.

3.3.4.3. LP/EUV
- LP ON.
- LP OFF.
- LP Deploy.

- EUV ON.
- EUV OFF.
- TBD.
- TBD.

3.3.4.4. VMAG

- ON.
- OFF.
- Deploy.
- Calibrate.

4. Nominal Spacecraft Operations

4.1. Propulsion Sequences

4.1.1. Trajectory Correction Maneuvers

4.1.1.1. All trajectory correction maneuvers will be done line-of-sight with the command uplink station.

4.1.1.2. Engine Selection
- TBD
- TBD

4.1.1.3. Pre-burn Sequence
- TBD
- TBD

4.1.1.4. Burn Sequence
- TBD
- TBD

4.1.1.5. Post-burn Sequence
- TBD
- TBD

4.1.2. Mars Orbit Insertion Maneuver

4.1.2.1. Engine Selection
- TBD
- TBD

4.1.2.2. Pre-burn Sequence
- TBD
- TBD

4.1.2.3. Burn Sequence
- TBD
- TBD

4.1.2.4. Post-burn Sequence
- TBD
- TBD

4.1.3. Mars Orbit Maneuvers

4.1.3.1. Engine Selection
- TBD
- TBD

4.1.3.2. Pre-burn Sequence
- TBD
- TBD

4.1.3.3. Burn Sequence
- TBD
- TBD

4.1.3.4. Post-burn Sequence
- TBD
- TBD

4.1.4. Spacecraft Reorientation

4.1.4.1. All S/C reorientation maneuvers will be done line-of-sight with a command uplink station.

4.1.4.2. Engine Selection
- TBD
- TBD

4.1.4.3. Pre-burn Sequence
- TBD

- TBD

4.1.4.4. Burn Sequence
- TBD
- TBD

4.1.4.5. Post-burn Sequence
- TBD
- TBD

4.1.5. Spacecraft Spin Up/Down

4.1.5.1. All Spin Up/Down maneuvers will be done line-of-sight with a command uplink station.

4.1.5.2. Engine Selection
- TBD
- TBD

4.1.5.3. Pre-burn Sequence
- TBD
- TBD

4.1.5.4. Burn Sequence
- TBD
- TBD

4.1.5.5. Post-burn Sequence
- TBD
- TBD

4.2. Attitude Control

4.2.1. Spacecraft Reorientation (see 4.1.4.).

4.2.2. Spacecraft Spin Up/Down (see 4.1.5.).

4.2.3. Horizon Scanners

4.2.3.1. TBD
- TBD
- TBD

4.2.3.2. TBD
- TBD
- TBD

4.2.3.3. TBD
- TBD
- TBD

4.2.4. Sun Sensors

4.2.4.1. TBD
- TBD
- TBD

4.2.4.2. TBD
- TBD
- TBD

4.2.5. Star Scanners

4.2.5.1. TBD
- TBD
- TBD

4.2.5.2. TBD
- TBD
- TBD

4.2.6. Gyros

4.2.6.1. TBD
- TBD
- TBD

4.2.6.2. TBD
- TBD
- TBD

4.2.7. Reaction Wheel

4.2.7.1. TBD
- TBD
- TBD

4.2.7.2. TBD
- TBD
- TBD

4.2.8. ACS Thrusters

4.2.8.1. TBD
- TBD
- TBD

4.3. Deployment Sequences

4.3.1. High Gain Antenna (see 4.5.3.2.)

4.3.2. Magnetometer Boom

4.3.2.1. TBD
4.3.2.2. TBD

4.3.3. Langmuir Probes

4.3.3.1. TBD
4.3.3.2. TBD

4.4. Command and Data Management

4.4.1. IEU

4.4.1.1. TBD.
4.4.1.2. TBD.

4.4.2. Realtime Command Sequence

4.4.2.1. TBD.
4.4.2.2. TBD.
4.4.2.3. TBD.

4.4.3. Stored Command Sequence

4.4.3.1. TBD.
4.4.3.2. TBD.
4.4.3.3. TBD.

4.5. Communications

4.5.1. Transmitter Selection

4.5.1.1. TBD.

4.5.1.2. TBD.

4.5.2. Recievers

4.5.2.1. TBD.

4.5.2.2. TBD.

4.5.3. Antennas

4.5.3.1. High Gain Antenna deployment will be done line-of-sight with a command uplink station.

4.5.3.2. High Gain Antenna Deployment Sequence

- Enable squib drivers.
- Arm antenna release squib.
- Fire antenna release squib.
- Disable squib drivers.

4.5.3.3. High Antenna Utilization

- The high gain antenna is the primary downlink antenna.
- The high gain antenna is primarily used for transmitting the 10, 20 and 40 kbps, high power, science downlink signal when the S/C is in its nominal orientations and when the ground tracking stations are in its antenna pattern.
- The high gain antenna can be used to downlink the low power, TBD bps data rate.
- The high gain antenna is the primary uplink antenna.

4.5.3.4. Omni Antenna Utilization

- The omni antenna is the backup uplink antenna.
- The omni antenna is used for both uplink and downlink of the low power, TBD bps data rate at all times the S/C is not in its nominal orientation.

4.6. Electrical Power

4.6.1. **Electrical power usage will never cause the batteries to discharge greater than 60% depth of discharge.**

4.6.2. **Integrated Bus Electronics**

4.6.2.1. TBD
- TBD.
- TBD.

4.6.2.2. TBD

4.6.3. **Solar Array**

4.6.3.1. TBD
- TBD.
- TBD.

4.6.3.2. TBD

4.6.4. **Power Distribution Electronics**

4.6.4.1. TBD
- TBD.
- TBD.

4.6.4.2. TBD

4.6.5. **Pyro Control Unit**

4.6.5.1. TBD
- TBD.
- TBD.

4.6.5.2. TBD

5. Check-Out Proceedures

6. Non-Nominal Operations

6.1. General Rules

6.1.1. **Non-Nominal Operations can be carried out only under the management of the on-duty Flight Director and the Mission Director.**

6.1.2. **Rules 1.?.?.?. and 1.?.?.?. may be suspended if non-nominal Spacecraft conditions require immediate action to secure a fail-safe Spacecraft condition.**

6.1.3. **During non-critical, Non-Nominal Operations the Spacecraft will be put in the "Appropriate Fail-Safe Mode" while the anomalous conditions are being analyzed by the Spacecraft Engineering Support Team. The Appropriate Fail-Safe Mode is that deemed correct by the Spacecraft Engineering Support Team, the on-duty Flight Director and the Mission Director for that particular anomaly (also see 1.?.).**

3/4/94

MISSION TIMELINE

FOR

THE MUADEE SPACECRAFT PROGRAM

Prepared by :

A. Binder

TABLE OF CONTENTS

1.0 Introduction

Section 2.0 contains the timeline for the MUADEE mission. The timeline starts with the launch of MUADEE and continues through the end of the mission 1073 days after launch. The timeline gives the day, hour, and minute when each event occurs, a short description of the event, and the operational manpower and tracking requirements needed throughout the mission.

2.0 Mission Timeline

MAUDEE Mission Timeline 1.0

Jan. 20, 1994

Time day/hr/min	Event	Ops Req.
	LAUNCH / CRUISE PHASE	
000/00/00	Launch	3 shifts, Full Ops teams, for 7 day check out, Continous S/C tracking for nav. until 3 days after 1st TCM
000/00/27	S/C separation, Initiate 7 day systems checkout period, transmitting on Omni at low power and low data rate	
000/00/47	Deploy high gain antenna	
000/00/57	Open latch valves	
000/01/07	Despin S/C to 3 RPM	
000/01/17	Put S/C in cruise attitude with +Z in the north direction of the Mars transfer orbit	
000/01/40	Close latch valves	
000/01/50	Deploy VMAG boom	
000/02/00	Switch to high data rate	
000/02/10	Turn VMAG ON to collect cal data in geomagnetic field and interplantary space and for checkout	
000/02/20	Deploy LP booms	
000/02/30	Turn LP ON for checkout	

000/03/30	Turn LP OFF	
000/03/40	Turn RPA/IDM ON for checkout	
000/04/40	Turn RPA/IDM OFF	
000/05/50	Turn NMS ON for checkout	
000/06/50	Turn NMS OFF	
000/07/00	Turn IP ON for checkout	
000/08/00	Turn IP OFF	
000/08/10	Turn UVS ON for checkout	
000/09/10	Turn UVS OFF	
000/09/20	Turn FPI ON for checkout	
000/10/20	Turn FPI OFF	
007/00/00	Turn VMAG OFF, End of check out period and cal data coll.	
007/00/10	Switch to low data rate	3 shift passive monitoring
014/00/00	Uplink 1st TCM commands	8 hour shift, full ops team
014/04/00	Ist TCM	
014/08/00		3 shift passive monitoring for 3 days for nav.
017/00/00		No ops.
021/00/00		8 hour Nav and S/C monitoring
021/08/00		No ops

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028/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
028/00/10	Turn LP ON for checkout	
028/00/20	Turn RPA/IDM ON for checkout	
028/00/30	Turn NMS ON for checkout	
028/00/40	Turn IP ON for checkout	
028/00/50	Turn UVS ON for checkout	
028/01/00	Turn FPI ON for checkout	
028/01/10	Turn LP OFF	
028/01/20	Turn RPA/IDM OFF	
028/01/30	Turn NMS OFF	
028/01/40	Turn IP OFF	
028/01/50	Turn UVS OFF	
028/02/00	Turn FPI OFF	No Ops
035/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
035/00/10	Down link stored VMAG data	
035/08/00	Switch to low data rate	No ops
042/00/00		Weekly 8 hr Nav and S/C monitoring
042/08/00		No ops

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049/00/00		Weekly 8 hr Nav and S/C monitoring
049/08/00		No ops
056/00/00	Uplink 2nd TCM commands	8 hour shift, Full ops team
056/00/20	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	
056/00/30	Turn LP on for checkout	
056/00/40	Turn RPA/IDM ON for checkout	
056/00/50	Turn NMS ON for checkout	
056/01/00	Turn IP ON for checkout	
056/01/10	Turn UVS ON for checkout	
056/01/20	Turn FPI ON for checkout	
056/01/20	Turn LP OFF	
056/01/30	Turn RPA/IDM OFF	
056/01/40	Turn NMS OFF	
056/01/50	Turn IP OFF	
056/02/00	Turn UVS OFF	
056/02/10	Turn FPI OFF	
056/04/00	2nd TCM	
056/08/00		3 shift passive monitoring for 3 days for nav.
057/00/00		No ops

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063/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
063/00/10	Down link stored VMAG data	
063/08/00	Switch to low data rate	No ops
070/00/00		Weekly 8 hr Nav and S/C monitoring
070/08/00		No ops
077/00/00		Weekly 8 hr Nav and S/C monitoring
077/08/00		No ops
084/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
084/00/10	Turn LP on for checkout	
084/00/20	Turn RPA/IDM ON for checkout	
084/00/30	Turn NMS ON for checkout	
084/00/40	Turn IP ON for checkout	
084/00/50	Turn UVS ON for checkout	
084/01/00	Turn FPI ON for checkout	
084/01/10	Turn LP OFF	
084/01/20	Turn RPA/IDM OFF	
084/01/30	Turn NMS OFF	

084/01/40	Turn IP OFF	
084/01/50	Turn UVS OFF	
084/02/00	Turn FPI OFF	No Ops
091/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
091/00/10	Down link stored VMAG data	
091/08/00	Switch to low data rate	No ops
098/00/00		Weekly 8 hr Nav and S/C monitoring
098/08/00		No ops
105/00/00		Weekly 8 hr Nav and S/C monitoring
105/08/00		No ops
112/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
112/00/10	Turn LP on for checkout	
112/00/20	Turn RPA/IDM ON for checkout	
112/00/30	Turn NMS ON for checkout	
112/00/40	Turn IP ON for checkout	
112/00/50	Turn UVS ON for checkout	
112/01/00	Turn FPI ON for checkout	
112/01/10	Turn LP OFF	

112/01/20	Turn RPA/IDM OFF	
112/01/30	Turn NMS OFF	
112/01/40	Turn IP OFF	
112/01/50	Turn UVS OFF	
112/02/00	Turn FPI OFF	No Ops
119/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
119/00/10	Down link stored VMAG data	
119/08/00	Switch to low data rate	No ops
126/00/00		Weekly 8 hr Nav and S/C monitoring
126/08/00		No ops
133/00/00		Weekly 8 hr Nav and S/C monitoring
133/08/00		No ops
140/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
140/00/10	Turn LP on for checkout	
140/00/20	Turn RPA/IDM ON for checkout	
140/00/30	Turn NMS ON for checkout	

140/00/40	Turn IP ON for checkout	
140/00/50	Turn UVS ON for checkout	
140/01/00	Turn FPI ON for checkout	
140/01/10	Turn LP OFF	
140/01/20	Turn RPA/IDM OFF	
140/01/30	Turn NMS OFF	
140/01/40	Turn IP OFF	
140/01/50	Turn UVS OFF	
140/02/00	Turn FPI OFF	No Ops
147/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
147/00/10	Down link stored VMAG data	
147/08/00	Switch to low data rate	No ops
154/00/00		Weekly 8 hr Nav and S/C monitoring
154/08/00		No ops
161/00/00		Weekly 8 hr Nav and S/C monitoring
161/08/00		No Ops
168/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
168/00/10	Turn LP on for checkout	

A.B. Binder, LESC, Jan. 20, 1994

168/00/20	Turn RPA/IDM ON for checkout	
168/00/30	Turn NMS ON for checkout	
168/00/40	Turn IP ON for checkout	
168/00/50	Turn UVS ON for checkout	
168/01/00	Turn FPI ON for checkout	
168/01/10	Turn LP OFF	
168/01/20	Turn RPA/IDM OFF	
168/01/30	Turn NMS OFF	
168/01/40	Turn IP OFF	
168/01/50	Turn UVS OFF	
168/02/00	Turn FPI OFF	No Ops
175/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
175/00/10	Down link stored VMAG data	
175/08/00	Switch to low data rate	No ops
182/00/00		Weekly 8 hr Nav and S/C monitoring
182/08/00		No ops
189/00/00		Weekly 8 hr Nav and S/C monitoring
189/08/00		No ops

A.B. Binder, LESC, Jan. 20, 1994

196/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
196/00/10	Turn LP on for checkout	
196/00/20	Turn RPA/IDM ON for checkout	
196/00/30	Turn NMS ON for checkout	
196/00/40	Turn IP ON for checkout	
196/00/50	Turn UVS ON for checkout	
196/01/00	Turn FPI ON for checkout	
196/01/10	Turn LP OFF	
196/01/20	Turn RPA/IDM OFF	
196/01/30	Turn NMS OFF	
196/01/40	Turn IP OFF	
196/01/50	Turn UVS OFF	
196/02/00	Turn FPI OFF	No Ops
203/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
203/00/10	Down link stored VMAG data	
203/08/00	Switch to low data rate	No ops
210/00/00		Weekly 8 hr Nav and S/C monitoring
210/08/00		No ops

A.B. Binder, LESC, Jan. 20, 1994

217/00/00		Weekly 8 hr Nav and S/C monitoring
217/08/00		No ops
224/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
224/00/10	Turn LP on for checkout	
224/00/20	Turn RPA/IDM ON for checkout	
224/00/30	Turn NMS ON for checkout	
224/00/40	Turn IP ON for checkout	
224/00/50	Turn UVS ON for checkout	
224/01/00	Turn FPI ON for checkout	
224/01/10	Turn LP OFF	
224/01/20	Turn RPA/IDM OFF	
224/01/30	Turn NMS OFF	
224/01/40	Turn IP OFF	
224/01/50	Turn UVS OFF	
224/02/00	Turn FPI OFF	No Ops
231/00/00	Switch to high data rate	Weekly 8 hr Nav and S/C monitoring
231/00/10	Down link stored VMAG data	
231/08/00	Switch to low data rate	No ops

A.B. Binder, LESC, Jan. 20, 1994

238/00/00		Weekly 8 hr Nav and S/C monitoring
238/08/00		No ops
245/00/00		Weekly 8 hr Nav and S/C monitoring
245/08/00		No ops
252/00/00	Turn VMAG ON to collect cal data interplantary space and for checkout, store 7 days data	8 hour Nav and S/C monitoring and Sci check
252/00/10	Turn LP on for checkout	
252/00/20	Turn RPA/IDM ON for checkout	
252/00/30	Turn NMS ON for checkout	
252/00/40	Turn IP ON for checkout	
252/00/50	Turn UVS ON for checkout	
252/01/00	Turn FPI ON for checkout	
252/01/10	Turn LP OFF	
252/01/20	Turn RPA/IDM OFF	
252/01/30	Turn NMS OFF	
252/01/40	Turn IP OFF	
252/01/50	Turn UVS OFF	
252/02/00	Turn FPI OFF	No Ops
259/00/00	Uplink 3rd TCM commands	8 hour shift,

259/04/00	3rd TCM	full ops team
259/08/00		3 shift passive monitoring for 3 days for nav.
262/00/00		No ops.
266/00/00		Weekly 8 hr Nav and S/C monitoring
266/08/00		No ops
273/00/00		Weekly 8 hr Nav and S/C monitoring
273/08/00		No ops
280/00/00	Turn VMAG ON for checkout	8 hour Nav and S/C monitoring and Sci check
280/00/10	Turn LP on for checkout	
280/00/20	Turn RPA/IDM ON for checkout	
280/00/30	Turn NMS ON for checkout	
280/00/40	Turn IP ON for checkout	
280/00/50	Turn UVS ON for checkout	
280/01/00	Turn FPI ON for checkout	
280/01/10	Turn VMAG OFF	
280/01/20	Turn LP OFF	
280/01/30	Turn RPA/IDM OFF	

280/01/40	Turn NMS OFF	
280/01/50	Turn IP OFF	
280/02/00	Turn UVS OFF	
280/02/10	Turn FPI OFF	No Ops
287/00/00	Begin MOI preparations	3 shifts, Full Ops teams for next 153 days, Start Continuous Tracking

ARRIVAL AT MARS / MOI / INITIAL ORBITAL OPERATIONS

289/23/45	Start MOI burn to 1 day, 33,100 x 250 km orbit
290/00/00	PERIAPSIS
290/00/15	End MOI Burn
290/00/25	Put S/C in Cartwheel Orientation
290/00/50	Put in 1 RPO Mode
290/01/10	Turn VMAG ON
290/08/00	Turn UVS ON to make Full-Disk Apoapsis Scan
290/12/00	APOAPSIS, Turn UVS OFF
290/23/25	Turn NMS ON for warmup
290/23/45	Turn Sci. Instruments ON
291/00/00	PERIAPSIS
291/00/15	Turn Sci. Instruments OFF, but leave VMAG ON

291/00/25	Put in 5 RPM Mode
291/02/00	Start Transmitting Sci. Data
291/04/29	Stop Transmitting Sci. Data
291/07/50	Turn UVS ON to make Full-Disk Apoapsis Scan
291/11/50	Turn UVS OFF
291/12/00	APOAPSIS, Small Burn to drop periapsis to 130 km, P = 23h 53.5m
291/23/22	Turn NMS ON for warmup
291/23/42	Turn Sci. Instruments ON
291/23/57	PERIAPSIS
292/00/12	Turn Sci. Instruments OFF, but leave VMAG ON
292/00/22	Put in 1 RPO Mode
292/02/00	Start Transmitting Sci. Data
292/04/29	Stop Transmitting Sci. Data
292/07/44	Turn UVS ON to make Full-Disk Apoapsis Scan
292/11/44	Turn UVS OFF
292/11/54	APOAPSIS
292/23/20	Turn NMS ON for warmup
292/23/40	Turn Sci. Instruments ON
292/23/50	PERIAPSIS

A.B. Binder, LESC, Jan. 20, 1994

293/00/05	Turn Sci. Instruments OFF, but leave VMAG ON
293/00/15	Put in 5 RPM Mode
293/02/00	Start Transmitting Sci. Data
293/04/29	Stop Transmitting Sci. Data
293/07/37	Turn UVS ON to make Full-Disk Apoapsis Scan
293/11/37	Turn UVS OFF
293/11/47	APOAPSIS, Small Burn to drop periapsis to 120 km, P = 23h 53.0m
293/23/09	Turn NMS ON for warmup
293/23/29	Turn Sci. Instruments ON
293/23/44	PERIAPSIS
293/23/09	Turn Sci. Instruments OFF, but leave VMAG ON
294/00/10	Put in 1 RPO Mode
294/02/00	Start Transmitting Sci. Data
294/04/29	Stop Transmitting Sci. Data
294/07/30	Turn UVS ON to make Full-Disk Apoapsis Scan
294/11/30	Turn UVS OFF
294/11/40	APOAPSIS
294/23/02	Turn NMS ON for warmup
294/23/22	Turn Sci. Instruments ON

A.B. Binder, LESC, Jan. 20, 1994

294/23/37	PERIAPSIS
294/23/52	Turn Sci. Instruments OFF, but leave VMAG ON
295/00/02	Put in 5 RPM Mode
295/02/00	Start Transmitting Sci. Data
295/04/29	Stop Transmitting Sci. Data
295/07/23	Turn UVS ON to make Full-Disk Apoapsis Scan
295/11/23	Turn UVS OFF
295/11/33	APOAPSIS, Small Burn to drop periapsis to 115 km, P = 23h 52.8m

AEROBRAKING PHASE

295/22/55	Turn NMS ON for warmup
295/23/15	Turn Sci. Instruments ON
295/23/30	PERIAPSIS
295/23/45	Turn Sci. Instruments OFF, but leave VMAG ON
295/23/55	Put in 1 RPO Mode
296/02/00	Start Transmitting Sci. Data
296/04/29	Stop Transmitting Sci. Data
296/07/16	Turn UVS ON to make Full-Disk Apoapsis Scan
296/11/16	Turn UVS OFF
296/11/26	APOAPSIS

A.B. Binder, LESC, Jan. 20, 1994

Timeline continues approximately like this for next 144 days or so, i.e., during all the aerobraking until the apoapsis is at 2500 km altitude (period = 2h 44m). Note, because the orbital period decreases, the data transmission time also decreases since less VMAG data are acquired per orbit and because of power constraints, data can not be taken every orbit. Also, UVS apoapsis Full-Disk scans are not made when the apoapsis altitude drops below 3400 km. Downlink data rated drops from 20 kbps to 10 kbps on day 395. As the orbit precesses the spacecraft must be periodically reoriented to keep it in the correct Cartwheel orientation. There will be 3 shifts, full ops and continuous tracking during this entire period.

440/03/38	Turn Transmitter OFF (Stored Data Downlinked)	
440/04/06	APOAPSIS, Small Burn to raise periapsis to 130 km, Apoapsis = 2500 km, P = 2h 44m	End Continuous Ops and Tracking

LATITUDE SURVEY PHASE (TAKE DATA EVERY 4TH ORBIT)

440/04/48	Turn NMS ON for warm up	No Ops
440/05/08	Turn Sci. Instruments ON	
440/05/28	PERIAPSIS	
440/05/48	Turn Sci. Instruments OFF, but leave VMAG ON	
440/06/50	APOAPSIS	
440/07/52	Turn VMAG OFF	
440/08/12	PERIAPSIS	
440/09/34	APOAPSIS	

440/10/56	PERIAPSIS	
440/12/18	APOAPSIS	
440/13/40	PERIAPSIS	
440/15/02	APOAPSIS	
440/15/54	Turn NMS ON for warm up	
440/16/04	Turn Sci. Instruments ON	
440/16/24	PERIAPSIS	
440/16/44	Turn Sci. Instruments OFF, but leave VMAG ON	
440/17/46	APOAPSIS	
440/08/48	Turn VMAG OFF	
440/19/08	PERIAPSIS	
440/19/38	Turn Transmitter ON (Earth Occultation Egress)	Passive Ops team plus tracking team
440/20/22	APOAPSIS	
440/21/42	Turn Transmitter OFF (Earth Occultation Ingress)	
440/21/52	PERIAPSIS	
440/22/22	Turn Transmitter ON (Earth Occultation Egress)	
440/23/14	APOAPSIS	
441/00/26	Turn Transmitter OFF (Earth Occultation Ingress)	

A.B. Binder, LESC, Jan. 20, 1994

441/00/36	PERIAPSIS	
441/01/06	Turn Transmitter ON (Earth Occultation Egress)	
441/01/30	Turn Transmitter OFF (Stored Data Downlinked)	No Ops
441/01/58	APOAPSIS	
441/02/40	Turn NMS ON for warm up	
441/03/00	Turn Sci. Instruments ON	
441/03/20	PERIAPSIS	
441/03/40	Turn Sci. Instruments OFF, but leave VMAG ON	
441/04/42	APOAPSIS	
441/05/44	Turn VMAG OFF	
441/06/04	PERIAPSIS	
441/07/26	APOAPSIS	
441/08/48	PERIAPSIS	

Time line continues approximately like this for next 480 days or so, except for the approximately 40 day period around solar conjunction. During solar conjunction period uplink and downlink will not be possible, so the spacecraft will be put in a hibernation safe mode. As the orbit precesses the spacecraft must be reoriented about once every 20 days to keep it the correct Cartwheel orientation. Full ops teams for one shift during reorientation maneuver and uplink of commands.

560/00/00	Put S/C in Hibernation Mode for about 40 day duration of Solar Conjunction

A.B. Binder, LESC, Jan. 20, 1994

SOLAR CONJUNCTION (40 DAYS - No Ops)

600/00/00	Wake up S/C, check out all subsystems, continue science mission	

AEROBRAKING PHASE (TAKE DATA EVERY 4TH ORBIT)

921/04/06	APOAPSIS, small burn to lower periapsis to 115, km	3 Shifts, Full Ops Teams for next 30 days, Start Continuous Tracking
921/04/48	Turn NMS ON for warm up	
921/05/08	Turn Sci. Instruments ON	
921/05/28	PERIAPSIS	
921/05/48	Turn Sci. Instruments OFF, but leave VMAG ON	
921/06/50	APOAPSIS	
921/07/52	Turn VMAG OFF	
921/08/12	PERIAPSIS	
600/09/34	APOAPSIS	
921/10/56	PERIAPSIS	
921/12/18	APOAPSIS	
921/13/40	PERIAPSIS	
921/15/02	APOAPSIS	
921/15/54	Turn NMS ON for warm up	
921/16/04	Turn Sci. Instruments ON	

921/16/24	PERIAPSIS
921/16/44	Turn Sci. Instruments OFF, but leave VMAG ON
921/17/46	APOAPSIS
921/08/48	Turn VMAG OFF
921/19/08	PERIAPSIS
921/19/38	Turn Transmitter ON (Earth Occultation Egress)
921/20/22	APOAPSIS
921/21/42	Turn Transmitter OFF (Earth Occultation Ingress)
921/21/52	PERIAPSIS
921/22/22	Turn Transmitter ON (Earth Occultation Egress)
921/23/14	APOAPSIS
922/00/26	Turn Transmitter OFF (Earth Occultation Ingress)
922/00/36	PERIAPSIS
922/01/06	Turn Transmitter ON (Earth Occultation Egress)
922/01/30	Turn Transmitter OFF (Stored Data Downlinked)
922/01/58	APOAPSIS
922/02/40	Turn NMS ON for warm up
922/03/00	Turn Sci. Instruments ON

A.B. Binder, LESC, Jan. 20, 1994

922/03/20	PERIAPSIS	
922/03/40	Turn Sci. Instruments OFF, but leave VMAG ON	
922/04/42	APOAPSIS	
922/05/44	Turn VMAG OFF	
922/06/04	PERIAPSIS	
922/07/26	APOAPSIS	
922/08/48	PERIAPSIS	

Time line continues approximately like this for next 30 days or so, i.e., the apoapsis is at 250 km. As the orbit precesses the spacecraft must be reoriented about once every 5 to 20 days to keep it the correct Cartwheel orientation. Full ops teams for one shift during reorientation maneuver and uplink of commands.

DIURNAL SURVEY (TAKE DATA EVERY 4TH ORBIT)

952/23/30	APOAPSIS, small burn to raise periapsis to 250 km, 250 x 250 km circular orbit, P = 1h 52m	End continuous Ops and tracking
952/23/40	Turn NMS ON for warm up	No Ops
953/00/00	Turn Sci. Instruments ON	
953/01/52	Turn Sci. Instruments OFF	
953/09/00	Turn NMS ON for warm up	
953/09/20	Turn Sci. Instruments ON	
953/11/12	Turn Sci. Instruments OFF	

A.B. Binder, LESC, Jan. 20, 1994

953/16/28	Turn NMS ON for warm up	
953/16/48	Turn Sci. Instruments ON	
953/18/40	Turn Sci. Instruments OFF	
953/19/00	Turn Transmitter ON (Earth Occultation Egress)	Full Ops and Tracking
953/20/10	Turn Transmitter OFF (Earth Occultation Ingress)	
953/20/52	Turn Transmitter ON (Earth Occultation Egress)	
953/22/02	Turn Transmitter OFF (Earth Occultation Ingress)	
953/22/46	Turn Transmitter ON (Earth Occultation Egress)	
953/23/56	Turn Transmitter OFF (Stored Data Downlinked)	No Ops
953/23/56	Turn NMS ON for warm up	
953/00/16	Turn Sci. Instruments ON	
953/02/08	Turn Sci. Instruments OFF	

Time line continues approximately like this for next 120 days. As the orbit precesses the spacecraft must be reoriented about once every 5 days to keep it the correct Cartwheel orientation. Also, as the orbit altitude decays 200 km, two small burns will be made to perodically put the S/C back into its 250 km circular orbit. Full ops teams for one shift during reorientation and burn maneuvers and uplink of commands.

1072/18/00	Burn to put S/C into 250 x 1000 km quarantine orbit.	Full Ops and tracking

A.B. Binder, LESC, Jan. 20, 1994

1073/00/00	Put S/C in Hibernation Mode	End Ops and tracking

END OF MISSION

A.B. Binder, LESC, Jan. 20, 1994

3/4/94

INSTRUMENT DESCRIPTION DOCUMENTS

FOR

THE MUADEE SPACECRAFT PROGRAM

Prepared by : A. Binder

TABLE OF CONTENTS

1.0 Introduction

The following three sections contain the Instrument Description Documents (IDD) for the three instruments packages which makeup the MAUDEE science payload. These packages are the Neutral Atmosphere Package (NAP), the Scanning Imaging Package (SIP), and the Plasma Instrument Package (PIP). Each section contains the IDD for the instrument package itself, followed by an IDD for each of the sensors which are in the instrument package. For example, the Neutral Atmosphere Package has two sensors, the Fabry-Perot Interferometer sensor and the Neutral/Ion Mass Spectrometer sensor. The IDD for each of the three packages contains the information needed to describe the integrated package, including the common elements, i.e., the Data Processing Unit (DPU) and the Power Supply (PS). Each sensor IDD gives those data which are unique to that sensor.

2.0 Neutral Atmosphere Package IDD

Neutral Atmosphere Package (NAP)

Team Leader:	Dr. Timothy Killeen
Address:	Space Physics Research Lab. U. Michigan 2455 Hayward Ann Arbor, MI 48109-2143
Telephone:	313/747-3435 W 313/426-5904 H
Fax:	313/763-0437
Team Engineer:	Brian Kennedy
Address:	Space Physics Research Lab. U. Michigan 2455 Hayward Ann Arbor, MI 48103-2143
Telephone:	313/764-6561 W 313/994-5205 H
Fax:	313/763-0437

Total Mass (kg):	16.9
Common Elements (List):	
SIE (Power Supply, RS422 I/F, 80c86 DPU):	2.8
NMS Elements (List):	
Sensor/Electronics	3.7
FPI Elements (List):	
Electronics:	2.8
Sensor:	3.0
Telescopes (2, total):	4.6

Dimensions (cm):	
Common Elements (List):	
SIE:	15 x 15 x 12
NMS Elements (List):	
Sensor/Electronics:	25 x 37.4 x 19.1
FPI Elements (List):	
Electronics:	12 x 15 x 15
Sensor:	15 x 15 x 60

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March 4, 1994

Telescopes (2, each):	10 x 15 x 15.2
Power (w)/Time Period:	
Worst Case Peak:	45
Operational Peak:	45
Average ON:	20.1
Average Standby:	8.4
Electronics:	5.1 +
Heaters:	10
Coolers:	3
DPU Requirements (Total/Op-Sys):	
CPU Time (%):	50/5
CPU Processing (MIPS):	0.15/0.015
EEPROM (K):	140/16
RAM (K):	56/8
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n, +/- V):	n, +/-6
Operational Temperatures (^oC):	
Common Elements (List):	
SIE:	-20 to +40
NMS Elements (List):	
Electronics:	-20 to +40
Sensor	-60 to +100
FPI Elements (List):	
Electronics:	-20 to +40
Sensor:	+15 to +25
Telescopes:	-20 to +40
Survival Temperatures (^oC):	
Common Elements (List):	
SIE:	-30 to +60
NMS Elements (List):	
Electronics:	-40 to +80
Sensor	-100 to +300
FPI Elements (List):	
Electronics:	-30 to +60
Sensor:	-30 to +60
Telescopes:	-30 to +50
Thermal Radiator:	

Area (cm^2):	100
Clear Field of View:	2π
Total Data Rate (bps):	1440
Science:	1364
Engineering:	76
Telemetry Format:	
Max. Bit Error Rate:	
Science:	10^{-5}
Engineering:	10^{-5}
Commands:	
Words (#):	36
Word Size (Bits):	16
Rate (bps):	TBD
Attached Instrument Drawings (y/n):	y
Special Issues:	None

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March 4, 1994

Fabry-Perot Interferometer (FPI)

Leader:	Dr. Timothy Killeen
Address:	Space Physics Research Lab. U. Michigan 2455 Hayward Ann Arbor, MI 48109-2143
Telephone:	313/747-3435 W 313/426-5904 H
Fax:	313/763-0437
Engineer:	Brian Kennedy
Address:	Space Physics Research Lab. U. Michigan 2455 Hayward Ann Arbor, MI 48103-2143
Telephone:	313/764-6561 W 313/994-5205 H
Fax:	313/763-0437

Total Mass (kg):	10.4
Sensors:	3.0
Electronics:	2.8
2 Telescopes:	4.6
Cables:	
Dimensions (cm):	
Sensors:	15 x 15 x 60
Electronics:	15 x 15 x 12
2 Telescopes:	10 x 15 x 15.2
Power (w)/Time Period:	
Worst Case Peak:	
Operational Peak:	30/20 msec every 0.25 sec
Average ON:	8.1/0.25 sec
Average Standby:	3 / ?
Electronics:	5.1
Heaters:	10
Coolers:	3

A. B. Binder
March 4, 1994

DPU Requirements:	
CPU Time (%):	25
CPU Processing (MIPS):	0.075
EEPROM (K):	64
RAM (K):	32
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n, +/- V):	n, +/-6
Operational Temperatures (^oC):	
Detectors:	-100
Sensors:	+15 to +25
Electronics:	-20 to +40
Telescopes:	-20 to +40
Survival Temperatures (^oC):	
Sensors:	-30 to +60
Electronics:	-30 to +60
Telescopes:	-30 to +50
Thermal Radiator:	
Area (cm^2):	100
Clear FOV (^{o}, half angle):	2π
Total Data Rate (bps):	860
Science:	800
Engineering:	60
Telemetry Format:	
Useful Data Altitudes (km):	
Instrument ON Range:	
Measurement Altitudes:	60 to 200
Type of Data:	
Limb Scans (y/n):	y
Disks Scans (y/n):	n
In Situ (y/n):	n
Duty Cycle, Specify:	
Trans-Mars Cruise:	0

A. B. Binder
March 4, 1994

Mars Orbit:	30 to 100%
Time:	
Scan (sec):	
Sample/Integration (msec):	
Synchronization (msec):	10
Absolute (sec)	
Orbit Knowledge (3σ):	
Position (km):	1 (alt., along and cross track)
Velocity (m/s):	3
Max. Bit Error Rate:	
Science:	10^{-5}
Engineering:	10^{-5}
Commands:	
Words (#):	20
Word Size (Bits):	16 bit
Rate (bps):	<1
Mounting:	Telescopes only
Look Direction wrt S/C:	45^{o}, 135^{o} wrt spin or velocity vector
Alignment Uncertainty (^{o}, 3σ):	0.1
Knowledge (^{o}, 3σ):	0.01
Clear FOV (^{o}, half angle):	45
Co-alignment w/ Other Instrument, Specify:	
Fields of View (^{o}):	
Direction:	45,135 wrt RAM/velocity vector
Instantaneous:	0.2 (V) x 1 (H)
Pointing on Orbit (3σ):	
Placement (^{o}):	1 R,P,Y
Knowledge (^{o}):	0.3 R,P,Y
Jitter (^{o}, sec):	0.3 in 0.25 sec
Stability (^{o}, sec):	0.3 in 10 sec
Instrument Produced Torques:	
Magnitude (Nm):	1.6×10^{-5}
Moment of inertia (g cm^2):	

A. B. Binder
March 4, 1994

Min. Rate of Occurance (sec):	0.25 sec
Duration (msec):	20

Deployment/Initial Turn On Sequence:	Power on, Telescope cover deploy, Calibrate, Operate
Operation Modes:	Calibrate, Limb scans
Rotation Rate (rpm):	TBD

Contamination Limits:

Magnetic (γ):	<50,000
S/C Potential:	N/A
Particulate (size & #/cm2):	TBD
Molecular (Angstroms):	TBD

Purges:	Dry N_2 purges of interferometer and telescopes
Attached Instrument Drawings (y/n):	n
Special Issues:	None

A. B. Binder
March 4, 1994

Neutral/Ion Mass Spectrometer (NMS)

Team Leader:	Dr. Hasso B. Niemann (alternate, Dr. Paul Mahaffy)
Address:	Code 915 NASA/GSFC Greenbelt, MD 20771
Telephone:	301/286-8706 (8184) W 410/730-6859 H
Fax:	301/286-2630 (1756 after 10/1/93)
Team Engineer:	Jack E. Richards
Address:	Code 915 NASA/GSFC Greenbelt, MD 20771
Telephone:	301/286-7250 W 301/262-2226 H
Fax:	301/286-2630 (1756 after 10/1/93)

Total Mass (kg):	3.7
Sensors:	1.8
Electronics:	1.2
Structure:	0.7
Cables:	
Dimensions (cm):	25 x 37.4 x 19.1 See drawings
Power (w)/Time Period:	
Worst Case Peak:	15
Operational Peak:	Stepping from 9 to 15 w over couple sec
Average ON:	12.0
Average Standby:	5.4
DPU Requirements:	
CPU Time (%):	20
CPU Processing (MIPS):	0.06
EEPROM (K):	60

A. B. Binder
March 4, 1994

RAM (K):	16
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n, +/- V):	y, +/-1
Operational Temperatures (^{o}C):	
Detectors:	
Sensors:	-60 to +100
Electronics:	-20 to +40
Survival Temperatures (^{o}C):	
Sensors:	-100 to +300
Electronics:	-40 to +80
Total Data Rate (bps):	580
Science:	564
Engineering:	16
Telemetry Format:	
Useful Data Altitudes (km):	
Instrument ON Range:	<500
Measurement Altitudes:	<500
Type of Data:	
Limb Scans (y/n):	n
Disks Scans (y/n):	n
In Situ (y/n):	y
Duty Cycle:	
Trans-Mars Cruise:	None
Mars Orbit:	All
Time:	
Scan (sec):	
Sample/Integration (msec):	
Synchronization (msec):	1.0
Absolute (sec):	
Orbit Knowledge (3σ):	
Position (km):	+/-5
Velocity (m/s):	0.2

A. B. Binder
March 4, 1994

Max. Bit Error Rate:	
Science:	10^{-5}
Engineering:	10^{-3}
Commands:	
Words (#):	16
Word Size (bits):	16
Rate (bps):	TBD
Mounting:	
Look Direction wrt S/C:	RAM direction preferred (if spinner, instrument boresight needs to point into RAM direction once per spin period)
Alignment Uncertainty (^{o}, 3σ):	0.1
Knowledge (^{o}, 3σ):	0.1
Clear FOV(^{o}, half angle):	??
Co-alignment w/ Other Instruments, Specify:	
Fields of View (^{o}):	
Direction:	Instrument boresight in RAM direction
Instantaneous:	2π Steradians
Pointing on Orbit:	
Placement (^{o}):	0.2
Knowledge (^{o}):	0.2
Jitter (^{o}, sec):	
Stability (^{o}, sec):	?
Deployment/Initial Turn On Sequence:	Break off cap - ejected after Mars orbit insertion/after orbit insertion no special turn-on sequence requirements
Operation Modes:	Neutral Open Neutral Closed Ion
Rotation Rate (rpm):	5

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March 4, 1994

Contamination Limits:

Magnetic (γ): $<5 \times 10^4$

S/C Potential (v): <5

Ground Test: Routine instrument tests

Attached Instrument Drawings (y/n): y

Special Issues:

Thruster contamination. Thruster location and plume direction.

A. B. Binder
March 4, 1994

3.0 Scanning Imaging Package IDD

Scanning Imaging Package (SIP)

Leader:	Dr. A. Ian F. Stewart
Address:	LASP, Campus Box 392 U. Colorado @ Boulder Boulder, CO 80309-0392
Telephone:	303/492-4630 or 8689 W 303/444-1330 H
Fax:	303/492-6444 or 6946
Engineer:	Sam Jones
Address:	LASP, Campus Box 392 U. Colorado @ Boulder Boulder, CO 80309-0392
Telephone:	303/492-6179 W H
Fax:	303/492-6444

Total Mass (kg):	13.8
Common Elements (List):	
DPU and Power Conversion Unit:	4.6
Harnesses:	TBD
UVS Elements (total):	5.0
Spectrometer and Detector:	1.7
Electronics:	1.1
Scan Mirror:	0.5
Slit Changer:	0.1
Telescope and Structure:	1.6
IP Elements (total):	4.18
Sensors:	4.03
Cables:	0.15

Dimensions (cm):	
Common Elements (List):	
DPU and Power Conversion Unit:	20.3 x 25.4 x 12.7
UVS Elements (List):	
UVS:	43.4 x 26.9 x 10.9

A. B. Binder
March 4, 1994

Baffle:	TBD
IP Elements (List):	
Sensors:	54 x 49 x 33.6
Electronics:	20 x 15 x 7.5
Cooler:	10 dia. x 8
Power (w)/Time Period:	
Worst Case Peak:	12
Operational Peak:	7.5
Average ON:	5.5
Average Standby:	1 (DPU only mode)
Electronics:	5.5
Heaters:	0 - but need thermal model for confirmation
Coolers:	0
DPU Requirements (Total/Op-Sys):	
CPU Time (%):	45/5
CPU Processing (MIPS):	0.135/0/015
EEPROM (K):	64/16
RAM (K):	32/8
DC Voltages:	
S/C supplied (V):	+28 +/-6
Regulated (y/n, +/- V):	y, +5, +/-15, +24
Operational Temperatures (°C):	
Common Elements (List):	
DPU:	-40 to +55
Individual Elements (List):	See UVS and IP IDDs
Survival Temperatures (°C):	
Common Elements (List):	
DPU:	-55 to +85
Individual Elements (List):	See UVS and IP IDDs
Thermal Radiator:	
Area (cm^2):	314 (radius = 10 cm) for IP
Clear Field of View (°):	150 exclude view of planet
Total Data Rate (bps):	1226
Science:	1156
Engineering:	70

A. B. Binder
March 4, 1994

Telemetry Format:	270 16 bit words per spin + IP telemetry
Max. Bit Error Rate:	
Science:	10^{-7}
Engineering:	10^{-7}
Commands:	
UVS	
Words (#):	5 words 4 times per orbit
Word Size (Bits):1	16
Rate (bps):	<0.05 bps average
IP	
Words (#):	3
Word Size (Bits):1	16
Rate (bps):	TBD
Attached Instrument Drawings (y/n):	y
Special Issues:	None

A. B. Binder
March 4, 1994

Ultraviolet Spectrometer (UVS)

Leader:	Dr. A. Ian F. Stewart
Address:	LASP, Campus Box 392 U. Colorado @ Boulder Boulder, CO 80309-0392
Telephone:	303/492-4630 or 8689 W 303/444-1330 H
Fax:	303/492-6444 or 6946
Engineer:	Sam Jones
Address:	LASP, Campus Box 392 U. Colorado @ Boulder Boulder, CO 80309-0392
Telephone:	303/492-6179 W H
Fax:	303/492-6444

Total Mass (kg):	5.0
Spectrograph & Detector:	1.7
Electronics (μP & Motor Drive):	1.1
Scan Mirror:	0.5
Slit Changer:	0.1
Telescope & Structure:	1.6
Cables:	In above

Dimensions (cm): See attached drawings

Power (w)/Time Period:	
Worst Case Peak:	7 (3w for inst. & 2w each for scan mirror and slit shanger)
Operational Peak:	5/2.0 w step of <0.25 s for each scan mirror step, 1 step/12 sec; 2.0 w step of <0.25 s for each slit change, 2 changes/orbit
Average ON:	3
Average Standby:	0, Standby is OFF
Electronics:	3

A. B. Binder
March 4, 1994

DPU Requirements:

CPU Time (%):	30
CPU Processing (MIPS):	0.09
EEPROM (K):	24
RAM (K):	16

DC Voltages:

S/C supplied (V):	+28 DC to run DC to DC converters
Regulated (y/n, +/- V):	y, +5, +/-15, +24

Operational Temperatures (°C):

Detectors:	-20 to +40
Electronics:	-20 to +40

Survival Temperatures (°C):

Detectors:	-45 to +55
Electronics:	-45 to +55

Total Data Rate (bps):	360 (@ % rpm, rate proportional to spin rate)
Science:	340
Engineering:	20

Telemetry Format:	270 16 bit words per spin (255 sci and 15 eng)

Useful Data Altitudes (km):

Spin-scan Imaging:	>3400
Limb Profiles:	<3400
Disk Scans:	All altitudes

Type of Data:

Limb Scans (y/n):	y
Disks Scans (y/n):	y
In Situ (y/n):	n
Other:	Spin-scan Imaging, build up from disk scans Sky Surveys, similar to disk scans

Duty Cycle:

Trans-Mars Cruise:	8 hrs, once/month
Mars Orbit:	Elliptical orbit, 4 hr apoapsis image & limb scanning below 1000 km

A. B. Binder
March 4, 1994

	Circular orbit, 50% of orbit doing limb scans
Time:	
Scan (sec):	1 - 8
Sample/Integration (msec):	4 - 32
Synchronization (msec):	1
Absolute (sec):	0.1
Orbit Knowledge (3σ):	
Position (km):	1
Max. Bit Error Rate:	
Science:	10^{-7}
Engineering:	10^{-7}
Commands:	
Words (#):	5
Word Size (Bits):	16
Rate (bps):	<0.05 bps, 4 x per orbit
Mounting:	
Look Direction wrt S/C:	See attached diagram. The UVS should be mounted in such a way that, with the scanning mirror in its central position, the look direction is perpendicular to the S/C spin axis. The instrument slits are coplanar with the S/C spin axis. The scan mirror motion deflects the line-of-sight within a range of 90 +/- 30° from the spin axis. The optical axis of the monochromator makes an angle of 60° with the spin axis. To reduce scattered-light and aerodynamic "vehicle glow" problems, it is desirable to recess the instrument into the S/C body.
Alignment Uncertainty (°, 3σ):	0.1 on all axes
Knowledge (°, 3σ):	0.1 on all axes
Clear FOV (°, half angle)	60 along spin axis 30 perpendicular to spin axis
Co-alignment w/ Other Instrument, Specify:	TBD

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March 4, 1994

Fields of View (^{o}):	
Direction:	90 +/-30 from spin sxis
Instantaneous:	1.38 x 0.46 (long axis coplanar with S/C spin axis
Pointing on Orbit (3σ):	
Placement (^{o}):	1.0
Knowledge (^{o}):	0.1
Jitter (^{o}, sec):	0.03
Stability (^{o}, sec):	0.1 per spin (nominally 12 sec)
Instrument Produced Torques:	
Magnitude (Nm):	$<2 \times 10^{-6}$
Moment of Inertia (g cm^2):	667
Min. Rate of Occurance (sec):	12, once per spin period
Duration (msec):	200
Deployment/Initial Turn On Sequence:	No special deployment or turn on sequences. Red tag dust covers are removed prior to launch
Operation Modes:	The monochromator and the scan mirror each have 2 modes. These mechanisms are independent of each other:
Monochromator:	Spectral mode: grating scans Wavelength mode: grating fixed
Scan Mirror:	Imaging mode: mirror scans +/- 15^o Stare mode: mirror is fixed
Rotation Rate (rpm):	5 nominal, 3 - 10 acceptable
Contamination Limits:	
Magnetic (γ):	<50,000
S/C Potential:	N/A
Particulate (size & #/cm2):	TBD, instrument handling, while mirrors are exposed to external environment, should be done in Class 10,000 or better clean environment.
Molecular (Angstroms):	<300 Ang.

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March 4, 1994

Ground Test:	Nominal aliveness testing will be required prior to launch.
Purges:	GN_2 desirable but not required
Attached Instrument Drawings (y/n):	y

Special Issues:
"MUVS refers to proposed MAUDEE Ultraviolet Spectrometer, "PVOUVS refers to existing Pioneer Venus Orbiter UVS. MUVS is based on PVOUVS, but incorporates the following modifications and upgrades:

1) Add protective sun sensor
2) Replace Cassegrain telescope w/ off-axis parabolic telescope
3) Wider monochromator slits
4) Upgrade grating drive
5) Add scanning mirror, drive and control
6) Add bistable entrance slit changer mechanism
7) Add ability to cycle grating through position program
8) Add ability to co-add internal data buffer loads
9) add ability to "window" buffer readout
10) modify integration period options
11) modify instrument status readout

For the TWO PRIME S/C OPTIONS (spinnerw/momentum wheel, or straight spinner), there will be NO CHANGES TO THE PROPOSED MUVS INSTRUMENT.

Instrument descope options:
1st descope: delete scanning mirror (5) and slit changer (6)
2nd descope: remove all other mods/upgrades EXCEPT SUN SENSOR (1)

A. B. Binder
March 4, 1994

Imaging Photometer (IP)

Leader:	Dr. Steve Mende
Address:	Lockheed 91-20, Bldg. 255 3251 Porter Dr. Palo Alto, CA 94304
Telephone:	415/424-3282 W 415/948-5623H
Fax:	415/424-3333
Engineer:	E. K. Aamodt
Address:	
Telephone:	415/424-3280W H
Fax:	415/424-3333

Total Mass (kg):	4.18
Sensors:	4.03
Electronics:	0
Cable:	0.15
Dimensions (cm):	
Sensors:	54 x 49 x 33.6
Electronics:	20 x 15 x 7.5
Cooler:	10 dia. x 8
Power (w)/Time Period:	
Worst Case Peak:	4 (<1 sec once per orbit)
Operational Peak:	1.5
Average ON:	1.5
Average Standby:	0 (standby is off)
Electronics:	1.5
Heaters:	0 - but need thermal model
Coolers:	0
DPU Requirements:	
CPU Time (%):	10

A. B. Binder
March 4, 1994

CPU Processing (MIPS):	0.03
EEPROM (K):	24
RAM (K):	8
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n, +/- V):	n
Operational Temperatures (^oC):	
Detectors:	-60
Sensors:	-20 to +40
Electronics:	-40 to +55
Survival Temperatures (^oC):	
Sensors:	-55 to +85
Electronics:	-55 to +85
Thermal Radiator:	
Area (cm^2):	314 (radius = 10 cm)
Clear FOV (^{o}, half angle):	75
Total Data Rate (bps):	866
Science:	816
Engineering:	50
Telemetry Format:	
Useful Data Altitudes (km):	
Instrument ON Range:	200 - 1000
Measurment Altitudes:	80 - 300
Type of Data:	
Limb Scans (y/n):	y
Disks Scans (y/n):	y
In Situ (y/n):	n
Other:	
Duty Cycle:	
Trans-Mars Cruise:	Calibration once/month
Mars Orbit:	100 % day time, few night time passes
Time:	
Scan (sec):	N/A

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March 4, 1994

Sample/Integration (msec):	4000
Synchronization (msec):	100
Absolute (sec):	1

Orbit Knowledge (3σ):

Position (km):	15

Max. Bit Error Rate:

Science:	10^{-7}
Engineering:	10^{-7}

Commands:

Words (#):	3
Word Size (bits):	16
Rate (bps):	TBD

Mounting:

Look Direction wrt S/C:	Look in orbit plane at limb, look at 100 km limb tangent alt.
Alignment Uncertainty (^{o}, 3σ):	
Knowledge (^{o}, 3σ):	0.1
Clear FOV (^{o}, half angle):	TBD
Co-alignment w/ Other Instruments, Specify:	

Fields of View (^{o}):

Direction:	-11.59 to -41.54
Instantaneous:	40 Horizontal, 4 Vertical

Pointing on Orbit (3σ):

Placement (^{o}):	
Knowledge (^{o}):	0.1
Jitter (^{o}, sec):	
Stability (^{o}, sec):	0.1/sec ??

Instrument Produced Torques:

Magnitude (Nm):	TBD
Moment of Inertia (g cm^2)	
Min. Rate of Occurance (sec):	Very infrequent adjustment to look angle, not during data acquisition
Duration (msec):	

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Deployment/Initial Turn On Sequence:	No high voltage device. Aperature door opening. Door avoids dust contamination, once in spaced door always open.
Operation Modes:	Normal operation with electronically programable wavelength selection, 6 simultaneous channels
Rotation Rate (rpm):	1/orbit
Contamination Limits:	
Particulate (size & #/cm2):	Standard clean room practices ??
Molecular (Angstroms):	Standard clean room practices ??
Ground Test:	Normal ground test flow. Calibration as late as possible.
Attached Instrument Drawings (y/n):	y
Special Issues:	None

A. B. Binder
March 4, 1994

4.0 Plasma Instrument Package IDD

Plasma Instrument Package (PIP)

Team Leader:	Dr. Rod Heelis
Address:	Center for Space Science U. Texas @ Dallas 2601 N. Floyd P.O. Box 830-688 F022 Richardson, TX 75083-0688
Telephone:	214/690-2822W H
Fax:	214/690-2761
Team Engineer:	C. R. Lippincott
Address:	Center for Space Science U. Texas @ Dallas 2601 N. Floyd P.O. Box 830-688 F022 Richardson, TX 75083-0688
Telephone:	214/690-2819 W H
Fax:	214/690-2761

Total Mass (kg):	12.55
Common Elements (List):	
Main Electronics and DPU:	9
Individual Elements (List):	
RPA/IDM	2.2
LP/EUV	0.5
VMAG	0.5
Sensor Aperature Plane:	0.35

Dimensions (cm):	
Common Elements (List):	
Main Electronics and DPU:	19 x 26.7 x 19
Individual Elements (List):	
RPA/IDM	14.6 x 17.4 x 24.4
LP	0.4 dia x 100 (Two)
EUV	8 dia x 5 (Two)

A. B. Binder
March 4, 1994

VMAG	8 x 8 x 10 (Two)
Sensor Aperature Plane:	0.13 x 40 x 28

Power (w)/Time Period:

Worst Case Peak:	
Operational Peak:	
Average ON:	10
Average Standby:	
Electronics:	
Heaters:	0.5 for boom mounted VMAG sensors

DPU Requirements (Total/Op-Sys):

CPU Time (%):	50/5
CPU Processing (MIPS):	0.15/0.015
EEPROM (K):	64/16
RAM (K):	41/8

DC Voltages:

S/C supplied (V):	28
Regulated (y/n, +/- V):	n

Operational Temperatures (^oC):

Common Elements (List):	0 to +40 All
Individual Elements (List):	
IDM/RPA	-10 to +50
LP/EUV	-100 to +250
VMAG	-20 to +40

Survival Temperatures (^oC):

Common Elements (List):	-30 to +60 All
Individual Elements (List):	
IDM/RPA	-30 to +60
LP/EUV	-100 to +250
VMAG	-30 to +60

Total Data Rate (bps):	1800
Science:	1650
Engineering:	150

Telemetry Format:

Max. Bit Error Rate:

Science:	10^{-6}

A. B. Binder
March 4, 1994

Engineering:	10^{-6}
Commands:	
Words (#):	4
Word Size (Bits):	32
Rate (bps):	Adaptable uplink
Attached Instrument Drawings (y/n):	n
Special Issues:	None

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March 4, 1994

Ion Drift Meter/Retarding Potential Analyser (IDM)

Team Leader:	Dr. Rod Heelis
Address:	Center for Space Science U. Texas @ Dallas 2601 N. Floyd P.O. Box 830-688 F022 Richardson, TX 75083-0688
Telephone:	214/690-2822W H
Fax:	214/690-2761
Team Engineer:	C. R. Lippincott
Address:	Center for Space Science U. Texas @ Dallas 2601 N. Floyd P.O. Box 830-688 F022 Richardson, TX 75083-0688
Telephone:	214/690-2819 W H
Fax:	214/690-2761

Total Mass (kg):	8.0
Sensors:	5.2
Cables:	TBD
Dimensions (cm):	See Diagrams
Power (w)/Time Period:	
Worst Case Peak:	
Operational Peak:	
Average ON:	8/100%
Average Standby;	
Electronics:	8
DPU Requirements:	
CPU Time (%):	20
CPU Processing (MIPS):	0.06

A. B. Binder
March 4, 1994

EEPROM (K):	20
RAM (K):	16
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n +/- V):	n, +/-6
Operational Temperatures (^oC):	
Sensors:	0 to +50
Survival Temperatures (^oC):	
Sensors:	-30 to +60
Total Data Rate (bps):	656
Science:	648
Engineering:	8
Telemetry Format:	
Useful Data Altitudes (km):	
Instrument ON Range:	<2000
Measurment Altitudes:	<2000
Type of Data:	
Limb Scans (y/n):	n
Disks Scans (y/n):	n
In Situ (y/n):	y
Duty Cycle:	
Trans-Mars Cruise:	Aliveness test once/month
Mars Orbit:	Maximize either side of periapsis
Time	
Scan (sec):	
Sample/Integration (msec):	
Synchronization (msec):	0.1
Absolute (sec):	
Orbit Knowledge (3σ):	
Position (km):	1.0
Velocity (m/s):	5
Max. Bit Error Rate:	

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March 4, 1994

Science:	10^{-5}
Engineering:	10^{-5}

Commands:	
Words (#):	1
Word Size (bits):	32
Rate (bps):	Adaptable uplink format

Mounting:	
Look Direction wrt S/C:	Sensor look directlon along RAM
Alignment Uncertainty (^{o}, 3σ):	1
Knowledge (^{o}, 3σ):	0.1
Clear FOV(^{o}, half angle)	60
Co-alignment w/ Other Instruments, Specify:	

Fields of View (^{o}):	
Direction:	Perpendicular to nadir, in orbit plane
Instantaneous:	90 cone

Pointing on Orbit (3σ):	
Placement (^{o})	0.5
Knowledge (^{o}):	0.1
Jitter (^{o}, sec):	<0.1 @ >0.1 Hz Amplitude
Stability (^{o}, sec):	<1/60 from nadir and in yaw

Deployment/Initial Turn On Sequence:	None
Operation Modes:	Mass Search RPA Scan Analyzer Scan
Rotation Rate (rpm):	Static instrument

Contamination Limits:	
Magnetic (γ):	<5000
S/C Potential:	Not driven by solar array potential
Ground Test:	GSE will be provided for bench test. Post-integration testing via S/C ground system.

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March 4, 1994

Attached Instrument Drawings
(y/n): y

Special Issues:

1) Maximize conducting area on RAM surface of spacecraft.
2) Insulate solar array from plasma (usually requires painting conducting interconnectors)
3) White glove handling
4) Oil free vacuum systems
5) Red tag cover removal before launch
6) Minimize microphonic vibrations at sensors
7) Caution near exposed grids

A. B. Binder
March 4, 1994

Langmuir Probe/Extreme Ultraviolet Sensor (LP)

Team Leader:	Dr. Larry Brace
Address:	Code 914 Bldg. 22, Rm G52 NASA/GSFC Greenbelt, MD 20771 OR SPRL 2455 Hayward Ann Arbor, MI 48109-2143
Telephone:	301/286-8575 (GSFC) W 410/956-2809 H
Fax:	301/286-1663
Team Engineer:	Charles Edmonson
Address:	SPRL 2455 Hayward Ann Arbor, MI 48109-2143
Telephone:	313/764-5148 W H
Fax:	313/763-5567

Total Mass (kg):	0.5
Sensors:	0.5 (Total for LP and SEUV)
Dimensions (cm):	
Sensors:	0.4 dia. x 100 (Two probes) 8 dia. x 5 (Two probes)
Electronics:	15 x15 x 15
Power (w)/Time Period:	
Worst Case Peak:	
Operational Peak:	
Average ON:	6 /25 % elliptical orbit 100% circular orbit
Average Standby;	

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March 4, 1994

Electronics:	4
DPU Requirements:	
CPU Time (%):	24
CPU Processing (MIPS):	0.072
EEPROM (K):	20
RAM (K):	16
DC Voltages:	
S/C supplied (V):	28
Regulated (y/n, +/- V):	n
Operational Temperatures (^oC):	
Sensors:	-100 to +250
Survival Temperatures (^oC):	
Sensors:	-100 to +250
Total Data Rate (bps):	580
Science:	500
Engineering:	80
Telemetry Format:	
Useful Data Altitudes (km):	
Instrument Range:	All, but 100 to 1000 most important
Measurment Altitudes:	All, but 100 to 1000 most important
Type of Data:	
Limb Scans (y/n):	n
Disks Scans (y/n):	n
In Situ (y/n):	y
Duty Cycle:	
Trans-Mars Cruise:	Daily Sun measurements
Mars Orbit:	10 to 30%
Time:	
Scan (sec):	1 (sweep voltages)
Sample/Integration (msec):	
Synchronization (msec):	1
Absolute (sec)	
Orbit Knowledge (3σ):	

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March 4, 1994

Position (km):	1 altitude, 10 along orbit
Max. Bit Error Rate:	
Science:	10^{-4}
Engineering:	
Commands:	
Words (#):	1
Word Size (bits):	16 or 32
Rate (bps):	
Mounting:	
Look Direction wrt S/C:	1 or 2 LPs perpendicular to velocity vector if despun, perpendicular to spin axis if spinning, one EUVS on top and one on botton of S/C
Alignment Uncertainty (^{o}, 3σ):	5
Knowledge (^{o}, 3σ):	5
Clear FOV (^{o}, half angle)	LP 45 free of obstacles in RAM Direction, EUV see the Sun
Fields of View:	
Direction (^{o}):	Axis of LP 90 to RAM direction
Pointing (3σ):	
Placement (^{o}):	5
Knowledge (^{o}):	5
Jitter (^{o}, sec):	<1/?
Stability (^{o}, sec):	
Deployment/Initial Turn On Sequence:	If possible deploy LP and turn on within 20,000 km of Earth, otherwise deployed after MOI, SEUV fixed.
Operation Modes:	Many, controlled by microprocessor (VPO few controlled by relay commands)
Rotation Rate (rpm):	1 to 10
Contamination Limits:	
S/C Potential (V):	0 to -2 (isolated solar array)

A. B. Binder
March 4, 1994

Attached Instrument Drawings
(y/n): n

Special Issues:
Boom length up to 1 m, time of deployment depends of S/C acceleration during MOI.
If only one EUV detector, it will be mounted on top of the S/C.
Battery sized to provide 1 full orbit in circular phase.

A. B. Binder
March 4, 1994

Vector Magnetometer (VMAG)

Team Leader:	Dr. Jim Slavin
Address:	Code 696 Bldg. 2, Rm 209 NASA/GSFC Greenbelt, MD 20771
Telephone:	301/286-5839 W 301/982-9221 H
Fax:	301/286-1648
Co-Team Leader:	Mario Acuna
Address:	Code 695 NASA/GSFC Greenbelt, MD 20771
Telephone:	301/286-7258 W H
Fax:	

Total Mass (kg):	0.5
Sensors:	0.5
Cables:	
Dimensions (cm):	
Sensors:	8 x 8 x 10 (2)
Electronics:	13 x 18 x 18
Power (w)/Time Period:	
Worst Case Peak:	
Operational Peak:	
Average ON:	3/100%
Average Standby;	
Electronics:	3/100%
Heaters:	0.25 for each boom mounted sensor package during solar occultation
DPU Requirements:	
CPU Time (%):	1

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March 4, 1994

CPU Processing (MIPS):	0.003
EEPROM (K):	8
RAM (K):	1
DC Voltages:	
S/C supplied (V):	TBD (Flexible)
Regulated (y/n, +/- V):	
Operational Temperatures (^oC):	
Sensors:	-20 to +40
Survival Temperatures (^oC):	
Sensors:	-30 to +60
Total Data Rate (bps):	580
Science:	
Engineering:	
Useful Data Altitudes (km):	All
Type of Data:	
Limb Scans (y/n):	n
Disks Scans (y/n):	n
In Situ (y/n):	y
Duty Cycle:	
Trans-Mars Cruise:	Several weeks, including the passage through the geomagnetic field
Mars Orbit:	100% desirable
Time:	
Scan (sec):	
Sample/Integration (msec):	
Synchronization (msec):	+/-0.1
Absolute(sec):	
Orbit Knowledge (3σ):	
Position (km):	+/-1
Max. Bit Error Rate:	
Science:	10^{-6}
Engineering:	10^{-6}

A. B. Binder
March 4, 1994

Commands:

Words (#):	20
Word Size (bits):	8
Rate (bps):	<1

Mounting:

Look Direction wrt S/C: +/-0.25° for sensors (optical cube provided for sensors)

Alignment Uncertainty (°, 3σ):	+/-0.25
Knowledge (°, 3σ):	+/-0.25

Pointing on Orbit (3σ):

Placement (°):	+/-0.3
Knowledge (°):	0.3
Jitter (°, sec):	+/-0.3

Deployment/Initial Turn On Sequence:	Magnetometers ON for boom deployment. Deployment while S/C still in geomagnetic field
Operation Modes:	Off, On, Calibration

Contamination Limits:

Magnetic (γ):	<0.1 DC, <0.03 AC @ 0 to 60 Hz
S/C Potential:	N/A
Particulate (size & #/cm2):	No visable contamination
Molecular (Angstroms):	No visable contamination

Ground Test:	EMI, Functional tests
Attached Instrument Drawings (y/n):	y

Special Issues:
See attachments. Boom length 6 m.

A. B. Binder
March 4, 1994

LMSC/F440063
14 May 1993

MUADEE

STRUCTURAL DESIGN

CRITERIA & ENVIRONMENTS

[INITIAL DRAFT]

**MARS UPPER ATMOSPHERE
DYNAMICS, ENERGETICS, AND EVOLUTION
PROJECT**

**PREPARED IN SUPPORT OF
THE STUDY BY
THE UNIVERSITY OF MICHIGAN
SPACE PHYSICS RESEARCH LABORATORY**

MUADEE

Lockheed Missiles & Space Company, Inc.
SPACE SYSTEMS DIVISION • SUNNYVALE, CALIFORNIA, 94088

FOREWORD

Structural design criteria are provided for the Mars Upper Atmosphere Dynamics, Energetics, and Evolution (MUADEE) mission extracted from the University of Michigan (U of M) applicable requirements, and expanded by internal Lockheed design criteria. The purpose of this document is to provide under a single cover requirements and criteria which cover structural design and verification. Specific structural criteria utilized by Lockheed and not detailed in U. of M. documentation are taken from the document LMSC/D887697B, "Space System Division Structural Design Criteria" dated 01 June 1992. This MUADEE specific document is formatted identically to the internal Lockheed criteria document. Additions, or significant modifications, to material in that document are noted with a bar in the left margin next to the changed or added paragraph. The U. of M. or other source of new material is noted immediately prior to the paragraph by an abbreviation of the source name. English units have been changed to SI units without notation.

Questions or comments concerning this document may be directed to R. W. Goldin, (408) 742-1901.

R. W. Goldin
Spacecraft Eng'g. Project Engr.

APPROVED BY:

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DOCUMENT CHANGE RECORD

Revision	Changes	Date
Draft		5/14/93

TABLE OF CONTENTS

1.0 INTRODUCTION

1.1 SCOPE

This document prescribes the structural and environmental engineering criteria for design, development and fabrication of the Mars Upper Atmosphere Dynamics, Energetics, and Evolution (MUADEE) spacecraft. These criteria are generated based upon the top level system requirements for the MUADEE mission as presented in the Mission Proposal (Appendix B, Ref. 61). The following are included:

- General requirements
- Design loads & environments
- Strength requirements
- Strength verification
- Structural criteria glossary
- Bibliography & references

This document does not cover any requirements for the design of Ground Support Equipment (GSE) These are provided primarily in the LMSC "Safety and Environmental Protection Standards" manual, Appendix B, Ref. 7. If the design of special GSE for this Program must utilize criteria different from, or not provided in, the Reference 7 document, such criteria will be reviewed with the U of M and the LMSC Occupational Safety & Health organization.

1.2 APPLICABLE DOCUMENTS AND REFERENCES

A partial bibliography of structural criteria related documents is contained in Appendix B. Documents specific to the MUADEE program are noted below They are referenced in this document by their short title and are not repeated in the Appendix. Where conflicts exist between this document and other criteria the following will be used as the order of precedence for which information to use:

Short title	Full Title
1) MUADEE Mission Proposal	(See Appendix B, Ref. 61)
2) (TBD)	

1.3 AUTHORITY

This document shall govern the structural and environmental engineering development of the MUADEE spacecraft components to ensure their structural adequacy and compatibility. The criteria defined herein are intended to embody the structural and environmental requirements specified by U of M and the NASA sponsoring center. This document is intended to fulfill the requirements for establishing design criteria and standardizing and controlling design practices.

1.4 SIGN CONVENTIONS , DEFINITIONS AND SYMBOLS

A right-hand, orthogonal, body-fixed XYZ reference coordinate system shall be used for the common spacecraft. The Z axis is through the center of the earth with +Z toward the earth; the -Y axis is along the orbit normal, with +Y toward the cold side; the X axis completes the coordinate system, with +X axis in the general direction of, but not always coincident with, the spacecraft velocity vector. Roll, pitch and yaw are defined as rotations about the X, Y, and Z axes respectively. The definitions and symbols used in the performance of the structural analyses shall be consistent with the Glossary definitions contained in Appendix A and the Symbols and Abbreviations delineated in Appendix B, Ref. 13.

2.0 GENERAL REQUIREMENTS

The structure shall be designed to provide strength, rigidity, and dynamic response characteristics as necessary to achieve the required system performance under the specified life-cycle environmental and operational conditions. In addition, consistent with structural design principles and assumptions listed herein, the structure shall be designed to achieve cost-effective minimum weight with due regard to total system costs, reliability and schedule constraints.

The MUADEE Spacecraft mechanical design shall accommodate the fields-of-view in the required viewing directions, volumetric, footprint, thermal radiator and mass requirements of the instrument complements.

The MUADEE Spacecraft design shall accommodate integration of the instruments onto the spacecraft in any order of instrument delivery. It shall also accommodate Mechanical Ground Support Equipment (MSGE) hard points and interfaces for operations such as lifting, rotating and transporting. All items to be installed, removed, or replaced at the spacecraft level shall be accessible without disassembly.
The Spacecraft shall accommodate mounting of the instruments with the spacecraft +x axis vertical.

The structural design shall be governed by critical flight conditions wherever possible. System design concepts should be selected so that the burden of accommodating non-flight loads and environments is borne by ground equipment, rather than flight hardware unless analysis shows burdening flight hardware to be cost effective. The structure shall not be designed to withstand loads, pressures, or environments due to system malfunctions that would in themselves result in failure to accomplish the mission, except where safety of personnel may be jeopardized.

No Single point failure in the spacecraft or instrument interface shall permanently preclude the spacecraft from supporting the mission science. A failure in one component or subsystem shall not induce failure into other parts of the spacecraft. The MUADEE spacecraft design shall support detection, isolation, and recovery capabilities for any single fault in the spacecraft to ensure the health and safety of the spacecraft. The spacecraft shall be capable of surviving the occurrence of any single failure for 24 hours. The spacecraft shall be designed to minimize the potential for orbital debris generation in both nominal operation and malfunction conditions.

Pertinent mass, physical, mechanical, thermal, and dimensional properties of the structure under all design conditions shall be defined and utilized in analysis. Metric Units shall be used unless design manufacturing heritage makes this impractical.

2.1 MASS PROPERTIES

2.1.1 Methods.- During the development, fabrication, and system test phases continuous refinement of structural, equipment and fluid weights, centers of gravity and mass moments of inertia shall be performed and results shall be provided to the design organizations. The mass property data shall be generated from statistical, empirical, analytical and actual test sources. The former will normally provide the basis for initial estimates and will be subsequently replaced by analytical and actual test data. Segmented mass property breakdowns matching the mass distributions of stress and structural dynamics finite element models shall be provided as required to those disciplines.

2.1.2 Design Mass Properties.- The Design Mass Properties are those which are to be used during the design phase and which are consistent with the design conditions being analyzed. The differences between the Design Mass Properties and the final "Actual" mass properties are dependent on: (1) the level of definition and design maturity at the time of final design analysis, and (2) the relationship of the design to the current state-of-the-art. To compensate for these known uncertainties "Contingency" weight shall be included in the determination of design mass properties as appropriate to minimize the likelihood that mass property changes later in the program will necessitate substantial re-analysis of the design.

2.1.3 Minimum Weight Design.- A primary goal of structural design/analysis, second only to satisfying strength and rigidity requirements, shall be to achieve cost-effective, minimum weight flight components/assemblies. Attention shall be directed during early phases of design to alternate methods of construction and attachment as well as the incorporation of advanced materials technology. During analysis and drawing release, attention shall be given to achieving positive margins of safety as close to zero as is practicable. When weight is a critical system parameter, mass properties data for original designs shall include estimates of minimum attainable weights as well as design weights for possible use where low weight becomes critical. The techniques or rationale required to provide the minimum weight configurations shall be documented in support of the design philosophy.

2.2 LIMIT LOADS & STRUCTURAL DYNAMICS

Limit loads and pressures, which are the maximums expected to act on a structure throughout its service life (see Appendix A, Load Types (1)), shall be determined for use in structural design and as a basis for definition of loads to be applied in structural proof, ultimate and life strength testing. They shall be determined for all structural elements for every potentially critical loading condition likely to be encountered in the life of the structure. They are utilized in combination with the applicable design factors specified in Section 2.4 to establish the structural design and test loads utilized in design/analyses and test verification of the structures.

The general requirements for determination of limit loads/pressures, and the associated structural dynamic response characteristics, are provided hereunder. Specific design load (and pressure) criteria which are applicable to most space vehicles/payloads with little or no change are delineated in Section 3.

2.2.1 Methods.- The determination of maximum expected (limit) structural loads, deflections, and dynamic environments shall be accomplished using proven analytical procedures. In cases where no reliable analytical procedures are available empirical data shall be used. These data shall be obtained from documents listed in Appendix B, from results of tests, or from any other relevant sources. The spectrum of loading environments for the flight system shall be evaluated to determine the critical combinations of anticipated limit load, pressure, and deflection conditions for design of the flight system.

2.2.2 Limit Loads & Deflections.- Design limit loads and deflections shall be determined from steady state and dynamic analysis using the environmental conditions in section 3.0. For initial design/analyses, quasi-static load factors shall be combined , when appropriate, with estimated random vibration responses to synthesize limit loads.

Limit loads shall be applied at the center of mass (CM) of functional components and subassemblies, configured for launch, to design their mounting interfaces. Limit loads shall be applied in one direction at a time and in such a way as to produce the maximum stresses.

2.2.3 System Natural Frequencies.- The structure shall be designed so that major assemblies and subassemblies have frequency response characteristics which will avoid adverse effects on design loads, and interaction with the control system(s) and equipment requiring stable positioning. Where such interaction criticality is likely to exist, early in the design phase design frequency constraints shall be defined for those critical structural assemblies in recognition of the fact that such criteria , rather than limit load conditions, may govern their structural design.

2.2.4 Equipment Support Structure Natural Frequencies.- Equipment support structure shall be designed to minimize the response of the equipment to mechanical shock, with due consideration for a possible need to attenuate acoustic/random-vibration induced response in certain frequency bands where particular equipment may be susceptible. The support structure should be designed to ensure a minimum frequency of 35 Hz for the combined system (see Section 2.2.6). Each separately mounted instrument component, configured for launch, shall have a fixed base frequency of 50 Hz. Each separately mounted instrument component with a mass of less than 25kg shall have a minimum fixed-base frequency of 100Hz. In the case of heavy items of equipment where this requirement would necessitate a very heavy structure, the supporting bracket shall be designed on the basis of the loads/accelerations resulting from the dynamic loads analyses. Components and equipment packages in this category shall be modeled in the spacecraft modal analysis, or loads shall be evaluated using shock spectra and component frequency data obtained from analysis or test (see also Section 3.4).

2.2.5 Acoustic And Random Vibration Environment.- Analyses shall be conducted to determine the random vibration and acoustic environment at all locations on or in the vehicle where these environments might influence functional or structural integrity. Empirical test results, if available, shall be used to supplement the analysis. Preliminary design shall include consideration of 99 percentile estimated random vibration levels. Design limit loads shall be based on acceptance test levels of vibro-acoustic inputs, and no structural degradation shall occur due to vibro-acoustic protoflight qualification testing. The results of vibro-acoustic analyses shall be supported by development test data if available. Loads determined from the foregoing shall be combined with those developed by vehicle/assembly accelerations experienced while the vibro-acoustic environment is present to comprise design limit loads and deflections.

2.2.6 Transient Vibration Environment.- All items with significant mass and natural frequencies of less than 50 Hz, or those that might participate strongly in major system modes shall be identified for inclusion in the loads dynamic model analysis, and the resulting dynamic loads shall be used in the design and test of the installation (see Section 2.2.4). Sine vibration shall be considered to simulate any estimated periodic mission environments or to satisfy other requirements (e.g. load, shock).

2.2.7 Deployed Natural Frequencies.- In order to avoid coupling with the vehicle control system the natural frequency (clamped at the spacecraft interface) for appendages in the deployed state shall be analyzed. Depending on the weight and frequency of the appendages it may be necessary to include such appendages in the all-up deployed dynamic math model of the spacecraft to enable assessment of their effect on the control system responses.

2.3 STRUCTURAL/MECHANICAL INTERFACES

The vehicle structure shall be physically and mechanically compatible with functional components and assemblies. The design shall account for: (1) direct physical interaction of vehicle structure and components, assemblies, and internal and external systems; and (2) indirect interaction of the vehicle with two or more components, assemblies or systems. When practicable, interface compatibility shall be verified by experiment.

2.4 DESIGN FACTORS

Factors of safety, and any other special multiplying factors used in design are identified. Considerations involved in establishing the values of all design factors shall be defined. Special factors relating to personnel safety, compatibility of materials, or type of construction shall be defined separately from the basic design factors of safety and applied to limit loads in the design where appropriate.

2.4.1 Limit Load Factors.- As previously defined, the limit load is the maximum load expected to act on a structure over its service life. Special factors applied to the "expected" limit load may be needed because of structural complexities and unknowns. These factors are delineated in the following paragraphs.

2.4.2 Dynamic Contingency Factors.- The Model Uncertainty Factor (MUF) for the elastic, dynamic analyses is a multiplying factor of 1.0 or greater that is applied to analytically derived loads, accelerations, clearance loss, etc., to compensate for the design maturity status of the launch vehicle forcing functions. Typical MUFs which may be established for a space vehicle, and which shall be applied to the elastic portion of the response loads and included as part of limit loads, are:

Preliminary Launch/Ascent Load Cycle (PLC)...... 1.50

Final Load Cycle (FLC).. 1.20

Verification Load Cycle (VLC).............................. 1.00

2.4.3 Factors Of Safety.- Basic design factors of safety (FOSs) as presented in Table 2-1 shall be used to account for uncertainties in the analysis of structures that cannot be defined in a rational manner. Analysis shall not be used to verify strength of elements fabricated from composite materials. The wider range of strength associated with composite structures must be taken into account by additional demonstrations such as development tests, proof tests and larger design factors (see Table 2-1 and Section 2.4.7). These FOSs are the required multiplying factors applied to limit load to establish structural integrity at both design yield and design ultimate strength levels. No FOS shall be applied to temperatures or temperature gradients. The "design temperatures," however, may include temperature uncertainty tolerances (See Section 2.6.3).

In all ground operations where personnel exposure is a consideration, the procedures set forth in Industrial Safety Code of California, as presented in the LMSC Safety and Environmental Protection Standards, Manual C-12 (Appendix B, Ref. 7), shall apply.

TABLE 2-1: FACTORS OF SAFETY

	EXPENDABLE LAUNCH VEHICLES (UNMANNED)				
METALLIC	YIELD	ULTIMATE	QUAL-TEST TO ULTIMATE	PROTO PROOF	**NOTES**
PROTO-QUAL	**1.25**	**1.40**	-	**1.25**	(1)
Analysis Only	**2.0**	**2.6**			(2)
COMPOSITE	YIELD	ULTIMATE	QUAL-TEST	PROOF/ PROTO	
PROTO-QUAL	**1.25**	**1.60**	-	**1.20**	(3)(4)(5)

<u>Notes:</u> (1) Reference MIL-HDBK-340 (Appendix B, Ref.46 Table VIII)
(2) Reference GSFC 420-05-02 (Appendix A)
(3) All composites should be proofed to at least 1.10 x limit load
(4) Additional factor of 1.33 shall be used in analysis (Refer to Section 2.4.7)
(5) Reference GSFC SPAR-3 (Appendix B, Ref.49 Sections 3.4.1.1 & 3.4.1.2)

2.4.4 Stress Concentration Factors.- The distribution of stress across a structural element is nominally uniform. Any irregularities, however, such as abrupt changes in section, notches, or holes cause stresses to increase locally at those points. These increases can be several times the nominal stress. Similarly, application of loading is localized in many cases, and stress patterns are disturbed by eccentricities or discontinuities. Stress concentration factors shall be applied, as appropriate, to account for these local stresses which are higher than that calculated by conventional engineering formulae. Also, when considering safe-life criteria (fatigue analysis), stress concentration factors shall be applied to limit load spectra. When considering ultimate or failure criteria in ductile structure for static or quasi-static loads, stress concentration factors shall not be included in the analysis because local yielding redistributes the load and reduces the localized high stresses.

2.4.5 Fitting Factor.- A fitting may be defined as any part used to transfer load at a joint from one load-carrying unit to another. A fitting factor of 1.15 shall apply to all portions of the fitting, the means of attachment (connections), and the bearing on the members joined. In the case of integral fittings, the part shall be treated as a fitting up to the point where the section properties become typical of the member.

2.4.6 Casting Factor.- Allowable mechanical properties for the cast material are based on minimum values obtained from tests of separately cast bars. The many variables inherent in the casting process, such as sensitivity to cooling rate and size, make it necessary to require an additional factor to be applied to structure made from castings. This is to insure that such structure adequately reflects the possibility of the material being under-strength. The pertinent multiplying factors which shall be applied are as follows:

1. Factor applied to yield load.......................... 1.15

2. Factor applied to ultimate load...................... 1.33

2.4.7 Composite Material Factor.- A material factor of 1.33 shall be applied to limit load for design of structural parts made from fiber reinforced composites. This factor is required to account for the observed discrepancy between allowable strengths and other material properties data based on material specification values for laminates and the observed strengths of production parts. This factor may be waived with the concurrence of the Program Design Engineering Manager, the Lead Stress Engineer and the U of M, provided applicable test data have been obtained. (See Appendix B, Ref. 15).

2.4.8 Weld Factor.- A material factor shall be applied to the ultimate strengths of welds. This reduction factor varies for different weld materials and heat treatments. Refer to Appendix B, Refs. 5 and 8 for specific recommended values.

2.4.9 Buckling Factor.- To account for unknown strains introduced by end conditions, structural imperfections, constructions, cutout, etc., a buckling reduction factor shall be applied to the predicted allowable buckling load. Specific values are presented in Appendix B, Refs. 1, 31, 32, and 33.

2.4.10 Pressure Loads.- Factors of safety, as applied to pressurized systems, are treated separately in Section 2.5.

2.4.11 Fatigue & Creep Life Factor.- Structures shall be designed to sustain four times the expected service-life spectrum of loads without experiencing failure or detrimental creep..

2.5 PRESSURIZED SYSTEMS

Design requirements for pressurized systems are delineated in three categories: (1) pressure vessels; (2) pressurized primary structure; and (3) liquid and pneumatic pressure systems. Pressurized systems may or may not include primary structure. A pressure vessel is a container designed primarily for pressurized storage of gases or liquids (Refer to the expanded definition in Appendix A, and in Appendix B, Ref. 24). Pressure vessels shall be designed in accordance with the requirements of one of the following specifications:

- MIL-STD-1522: Vessels used in Missile and Space Systems (Appendix B, Ref. 24).

- ASME Boiler and Pressure Vessel Code: Vessels used in Ground systems, and which do not contain aggressive fluids (Appendix B, Ref. 25).

Design Factors of Safety for pressurized systems are delineated in four categories: (1) pressurized primary structure, (2) pressure vessels, (3) liquid and pneumatic systems, and (4) sealed containers.

2.5.1 Pressurized Primary Structure.- Pressure vessels serving as primary structure shall have strength and rigidity to withstand the maximum forces or combination of forces resulting from:

1. The static, vibratory, and repeated loads for all design load conditions.

2. The minimum operating pressure as defined in Appendix A.

3. The limit pressure as defined in Appendix A.

4. The thermal loads for all thermal conditions.

The design factors of safety of Table 2-1 are applicable.

2.5.2 Pressure Vessels.- All pressure vessels shall be capable of withstanding the proof pressure (at operating temperature and static 1g loading conditions) specified in Table 2-2 without exceeding the allowable yield pressure. All pressure vessels shall be capable of withstanding the ultimate pressure (at operating temperature and static 1g loading conditions) specified in Table 2-2 without rupturing. All pressure vessels and their supporting structure shall withstand the limit pressure applied simultaneously with the limit accelerations/ load factors without exceeding the allowable yield load (or pressure). All pressure vessels and their supporting structure shall withstand the limit pressure applied simultaneously with the ultimate accelerations/load factors without failing.

2.5.3 Pressure Systems.- Liquid and pneumatic pressure systems shall be designed to meet the requirements of MIL-H-25475 (Appendix B, Ref. 19) and MIL-P-5518 (Appendix B, Ref. 20). Test requirements shall be in accordance with MIL-STD-1540 (Appendix B, Ref. 23). No part of the pressure system shall fail, or be damaged in any manner, when subjected to the proof pressure specified in Table 2-3, and no part of the pressure system shall rupture when subjected to the ultimate pressure specified in Table 2-3. System relief pressures and peak pressure transients shall not exceed the values as specified in MIL-H-25475 and MIL-P-5518.

2.5.4 Sealed Containers.- Sealed containers shall be analyzed to establish hazard potential. Containers with hazardous potential shall be proof tested to 1.5 times the nominal pressure differential.

2.5.5 Vented Containers.- Venting provisions in vented containers shall be designed in accordance with the requirements of Section 2.9. The vent path of a piece of equipment will not directly impinge on any instrument's contamination-sensitive surface nor directly enter another instrument's aperture. If the pressures developed during venting are considered critical for design, the safety factors specified in Table 2-2 (1.) shall be applied to determine design pressures.

TABLE 2-2: TYPICAL PRESSURE VESSEL CRITERIA

CHARACTERISTICS	Factor Applied To Limit /MEOP Pressure (1)		
	Proof/Yield	Ultimate	App. B, Ref.
1. PRESSURE VESSELS, LIQUID AND GAS			
(a) For Conditions Hazardous to Personnel	1.5	2.0 (2)	21,24,46
(b) For Conditions Non-Hazardous to Personnel	1.2	1.6	22,24
2. MAIN LIQUID-PROPELLANT SUPPLY TANKS			
(a) For Conditions Hazardous to Personnel	1.5	2.0 (2)	22,24
(b) For Conditions Non-Hazardous to personnel	1.25	1.5	24
3. OTHER PRESSURIZED BOXES HAZARDOUS TO PERSONNEL	2.0	4.0	7,24
4. HEAT PIPE PRESSURE VESSEL			
(a) For Conditions Hazardous to Personnel	2.0	4.0	7,24
(b) For Conditions Non-Hazardous to personnel	1.5	2.5	-
(c) Collapse Pressure (external)	1.5	2.5	-

NOTES: (1) When not specified by U of M these criteria will be used. Refer to Appendix B, Refs. 7, 19, 20, 21, 22, 23, 24, 25, 42, 46, 49 and 50.

(2) These values require LMSC Industrial Safety Approval.

TABLE 2-3: TYPICAL PRESSURE SYSTEMS CRITERIA

CHARACTERISTICS	Factor Applied To Limit (MEOP) Pressure		App. B, Ref.
	Proof/Yield	Ultimate	
1. HYDRAULIC AND LIQUID-PROPELLANT SYSTEMS			19,22,23, 24
(a) Lines and Fittings diameter < 1.5 in.	1.5	4.0	
(b) Lines and Fittings diameter > 1.5 in.	1.5	2.5	
(c) Fluid Return Sections	1.5	3.0	
(d) Fluid Return Hose	1.5	5.0	
(e) Collapse Pressure of Parts Subject to Low or Negative Pressure	1.5	2.5 x maximum external pressure	
(f) Other Components (Valves, Filters, Transducers, etc.) Except Pressure Vessels	1.5	2.5	
(2) PNEUMATIC SYSTEMS			20,24
(a) Lines and Fittings (Mechanical and Brazed) and Hose	2.0	4.0	
(b) Actuating Cylinders Which Also Act as Reservoirs	2.0	4.0	
(c) Actuating Cylinders and Other Components	1.5	2.5	

NOTE: When comparable criteria are not specified by U of M these criteria will be used. Refer to Appendix B, References 7, 19, 20,21, 22, 23, 24, 25, 42, 46, 49 and 50.

2.6 THERMAL REQUIREMENTS

The MUADEE spacecraft thermal control design shall maintain all spacecraft subsystem , components and instrument interfaces at specified temperature levels, thermal gradients, and temperature transition rates consistent with a 3-year post-launch lifetime (Earth years).

2.6.1 Methods.- Temperature ranges, temperature differences, and thermal cycling shall be determined analytically in support of the evaluation of structural design adequacy, clearance margins and differential thermal expansion problems. Proven analytical techniques or empirical data shall be utilized. The thermal math model shall be formulated to enable assessment of worst case combinations of equipment operation, internal and external heat sources, vehicle orientation, eclipse conditions, and degradation of thermal control components, and to determine design temperatures (Refer to Appendix A and Section 2.6.3). Environments including prelaunch, ascent and free molecular heating, orbital heating from solar and earth effects, thermal shock due to earth shadow or self-shadowing effects, engine plume heating, and local strip heating of pressure vessels shall be considered wherever applicable. Particular attention shall be given to thermal analysis of joints, pivots and similar mechanisms where differential expansion of dissimilar materials could cause binding or loosening resulting in reduced design reliability or a potential safety hazard. Where thermal effects on structures are significant, worst case temperatures as determined using the thermal math model shall be utilized in design in combination with limit loads expected at the time of the most severe temperature conditions. For certain classes of problems, such as thermally induced dynamic response of solar arrays on S/C entering or exiting the earth's or Mars shadow, time histories of thermal temperatures and gradients will be required to assess loads and response.

2.6.2 Orbit Environment.- Mechanisms such as solar arrays and deployable structure shall be evaluated such that minimum and maximum temperatures and thermal gradients are determined for structural elements critical to the deployment or orbit articulation of the mechanism, or to induce dynamic disturbances to the vehicle. Thermal conditions for the various operational conditions, such as stowed, erected and deployed, shall be evaluated. Particular attention shall be placed on components subject to differential thermal expansion problems and such things as organics where blocking, stiffening or viscosity changes may result during extreme hot or cold conditions.

Misalignment of antennas and optical components due to structural thermal deformation shall be evaluated using temperature predictions resulting from orbital structural thermal analysis. These analyses shall consider extremes in orbital altitudes, time in earth or mars shadow, and environmental variations. Orbital thermal analyses shall use mathematical models which account for thermally extreme mission phases and environmental conditions, but only nominal structural physical characteristics. The combination of worst case predicted temperatures together with nominal structural properties shall be utilized in the evaluation of thermal distortions.

Heating of external spacecraft surfaces or appendages due to hot gas engines shall be considered when determining structural survivability. Particular attention shall be placed on antennas, solar booms or light weight structure. Deflections of these structures toward the plume centerline shall be evaluated when the engine firing itself will deflect the component. Where appropriate, high temperature shields or blankets shall be used to avoid direct plume impingement on critical structures.

Hot spots and resulting thermal gradients due to electrical heating, especially on pressure vessels, shall be evaluated in addition to the extreme temperature levels and differences which may exist during ascent, descent, orbit or depressurization. Hot spots shall be minimized by application of good thermal design practices, including the use of low power density heaters and judicious placement of heaters relative to the location of control thermostats or thermistors.

2.6.3 Design Temperatures.- Maximum (and minimum) temperatures of equipment items, determined by use of the thermal math model for worst case conditions, shall be increased (reduced) by 11° C for worst case thermal design and for use in equipment item acceptance tests. Minimum temperatures for equipment items shall not be reduced if the equipment is provided with a minimum of 25% excess heater control authority. Maximum (and minimum) temperatures of equipment items shall be increased (reduced) by 11° C more than acceptance test limits for component qualification tests.

2.7 MATERIAL PROPERTIES

Values of allowable mechanical and physical properties shall be selected from authorized reference sources such as those listed in Appendix B. Where values for new materials or joints, or values for existing materials or joints in new environments, are not available, they shall be determined by accepted and approved analytical and test methods.

Where tests are required, they shall be in sufficient number to establish values for the mechanical properties on a statistical basis, and the tests shall conform to the procedures set forth in MIL-HDBK-5 (Appendix B, Ref. 13) for metallic materials, MIL-HDBK-17 (Appendix B, Ref. 14) for composite plastics, and MIL-HDBK-23 (Appendix B, Ref. 15) for composite sandwich materials. The cumulative effects of temperature, thermal cycling and gradients, and detrimental environments shall be taken into account in defining allowable mechanical properties.

2.7.1 Metallic Design Values.- Materials designated by "A", "B", or "S" values are defined in MIL-HDBK-5, Section 1.4. "A" and "B" values are probability values based on statistical evaluation of test results. Minimum mechanical property values specified by procurement and process specifications without having statistical assurance are classified "S" values.
For materials having established "A" and "B" values, the "A" values shall be used in all applications where failure of a single load path would result in loss of the system. If "A" properties are not available, "S" properties shall be used for analysis. Material "B" values may be used in multi-load path applications in which the failure of a member would result in a safe redistribution of applied loads to other load-carrying members.

2.7.2 Composite Materials.- Composite materials for which "A", "B", or "S" values have been established shall be treated in the same manner as metallics (See Section 2.7.1). Many composite materials, however, have not been characterized sufficiently to establish "A", "B", or "S" values and therefore the properties should be determined by test, and allowable strengths and stiffnesses established in concurrence with the Structures Group Engineer. See Section 2.4.7 for use of the Composite Material Factor.

2.7.3 Conditions And Environments.- The properties of materials proposed for use shall be investigated for compatibility with the specific conditions and environments specified for the space system. Potentially critical considerations may be gases generated from structural elements, dirt particles retained on structural elements, loss of material strength due to outgassing, deflections due to moisture loss, and corrosion effects.

Special consideration shall be given to materials not subject to stress corrosion when material selection is evaluated. MSFC-SPEC-522A (Appendix B, Ref. 26) discusses the problem of stress corrosion, and places a number of materials into three categories: (1) Alloys and tempers which by testing and experience have been shown to possess high resistance to stress corrosion cracking; (2) Alloys and tempers which have been shown to possess moderate resistance to stress corrosion cracking; and (3) Materials which have been found to be highly susceptible to stress corrosion cracking. Categories (2) and (3) must be substantiated by a "Material Usage Agreement" (MUA).

2.8 VEHICLE STABILITY

Free standing structures subject to overturning shall be investigated to verify stability of the structures and safety of personnel during such free standing mission operations. The mass-weight of the structure shall be capable of producing a stabilizing moment which is 50% greater than the overturning moment caused by external loading, such as is produced by winds and earthquakes.

2.9 VENTING PROVISIONS

Provisions shall be made to preclude over-pressurization damage to any part of the vehicle/payload during transportation, launch, ascent, on orbit and/or descent, as applicable. Venting requirements based on the need to achieve very low pressures (near vacuum, due to corona, surface contamination, etc. effects) may exceed the other requirements for venting, and therefore should be addressed early in the design process. Openings provided for venting shall be designed without screens, filters, or other obstructions whenever practicable, and shall be located in aerodynamically benign areas where external pressure fluctuations are at a minimum.

3.0 DESIGN LOADS & ENVIRONMENTS

Design limit loads shall be determined in accordance with the general requirements specified in Section 2.2. Under this Section 3 limit load criteria are specified for design conditions which are applicable to the MUADEE space vehicle/payload system. The spacecraft shall be of sufficient strength and stiffness to maintain structural integrity and withstand all ground testing, handling, transportation, launch and mission orbit environments presented in this section and Section 6.0.

3.1 GROUND HANDLING

Limit load criteria for handling and transport conditions are specified hereunder.

1. HOISTING.- The minimum vertical components of the hoisting load shall be 2.0w, where w is the maximum weight to be hoisted (payload plus AGE). A load of 1.5w may be used when the hoist is equipped with hydra-set.

2. CRANE RUNWAY FORCES.- The simultaneous acting forces on crane runways to provide for the effect of moving crane trolleys shall be a vertical load of 1.25w, a lateral load of 0.2w and a longitudinal load of 0.1w, where w is the sum of the weights of the lifted load and the crane trolley.

3. JACKING.- For redundant systems which maintain control of equipment when one jack fails, the minimum vertical components of the jacking loads shall be 1.5 times the static reactions at the jack points and the minimum horizontal components shall be 0.15 times the static reactions, applied at the top of the jacks. For systems in which the failure of one jack results in loss of control of the equipment, the minimum vertical components of the jacking loads shall be 2.0 times the static reactions at the jack points, and the minimum horizontal component shall be 0.5 times the static reaction.

4. TOWING AND MOVING.- Ground equipment which is mobile shall be designed for loads encountered in towing and moving. Special handling criteria shall apply to that equipment which is to be moved only on smooth floors, rails or runways at a velocity not to exceed 7.5 miles per hour. Smooth is defined as having no abrupt changes in pavement which exceed one twentieth (1/20) of the wheel diameter. Tow forces, and direction of tow forces and reactions shall be as specified in Table 3-1. Gravitational force components due to towing up inclines shall be added to the values of Table 3-1. For moving conditions the inertia factors at the C.G. of the equipment shall be as specified in Table 3-1. The inertia forces shall be reacted by forces at the wheels, casters or air bearing pads as specified in Table 3-1.

5. BRAKING.- The vertical load factors at the C.G. shall be 1.0. A drag reaction at each wheel equipped with brakes shall be assumed acting at the ground and equal to the static coefficient of friction of the wheel on the ground times the vertical reaction on the wheel. For equipment restricted to special handling criteria, the drag reaction at each wheel shall be 40% of the wheel vertical reaction.

6. GROUND AND AIR TRANSPORTATION.- The transportation loads are the steady, shock, and vibratory loads encountered during transportation. The environments specified herein define the container's external loading environment. The dynamic environmental levels specified (i.e., shock and vibratory) are the extremes expected at the container/transporting vehicle interface. These levels are shown in Tables 3-2, 3-3, and 3-4. C5A aircraft low frequency design data are given in Figure 5, Appendix B, Ref. 38 and Figures 8, 9, and 10, of Ref. 39.

Containers shall be designed to withstand and protect their contents from appropriate drop and impact tests prescribed in Table 3-2. The type and severity of the environment to which the containers must be designed depends on the gross weight and length of the containers. Normally, shock isolator systems shall be designed to protect the space vehicle components and assemblies with respect to shock conditions under those conditions which are more than those experienced by the spacecraft during its applicable launch, ascent, and on-orbit conditions (See also Section 3.6-7).

7. SEISMIC LOAD FACTORS.- In California, and in other regions where significant seismic activity occurs, all equipment must withstand a limit load factor of 0.5 due to earthquake forces. The load is applied at the equipment C. G.. and is assumed to come from any horizontal direction.

8. AMBIENT PRESSURES.- The effects of ambient pressures, and ambient pressure changes including rapid decompression/recompression, shall be considered for superposition on other applicable load conditions for possible significance in regard to structural adequacy. (Refer also to Section 3.5).

TABLE 3-1: LIMIT TOWING AND MOVING CONDITIONS FOR MOBILE SUPPORT EQUIPMENT

Conditions	Mobility Description	Rolls on Rails		Wheels or caster - Front Swivel		Wheels or caster - All Swivel		Air Pads
		Special Handling	Ground Handling	Special Handling	Ground Handling	Special Handling	Ground Handling	Special Handling
Towing Fore & Aft	Tow Force (lb)	.15 W		$W(.2 + \frac{2000}{W + 10000})$				0.10W
	Force Direction	Parallel to rails ± 10°		Fore & Aft ± 30°	Fore & Aft ± 30°	All	All	All
	Reactions a.	Body Inertia						Body Inertia
	or b	at wheels						at pads
Towing Sideward	Tow Force (lb)			$W(.1 + \frac{1000}{W + 10000})$	$W(.1 + \frac{1000}{W + 10000})$			0.10W
	Force Direction			Normal to ℄	Normal to ℄			Normal to ℄
	Reactions a.	Body Inertia						Body Inertia
	or b	At wheels						At Pads
Moving Fore & Aft Acceleration (G's)	Nz (Vert.)	1.25		2.0	$2 + \frac{2000}{W + 10000}$	2.0	$2 + \frac{2000}{W + 10000}$	1.10
	Nx (Drag)	.15	.25	0.25	0.5	0.25	0.5	0.25
	Reactions -Vert	On all wheels to maintain equilibrium						
	Drag	On two wheels						
Moving Lateral Acceleration (G's)	Nz (Vert.)	1.25		2.0	$2 + \frac{2000}{W + 10000}$	2.0	$2 + \frac{2000}{W + 10000}$	1.10
	Ny (Side)	.10	.25	.15	.25	.25	.50	0.25
	Reaction vert	On all wheels to maintain equilibrium						
	side	On two wheels						
	drag	0		On all wheels to balance yaw due to Ny				

TABLE 3-2: PARTS AND COMPONENTS, TRANSPORTATION/HANDLING SHOCK VIBRATION (LIMIT LOADS)

TRANSPORTATION SHOCK

	Impulse (g's)		
Gross Weight (lb)	Rail	Truck	Aircraft
0 to 250	65	10	8
250 to 500	45	8	6
Duration - Milliseconds	10 - 50	8 - 40	0.8 - 40

TRUCK VIBRATION

Acceleration, 0-Peak (g)	Frequency (Hz)
3	2 - 10
4	10 - 100
6	100 - 1000

HANDLING SHOCK

Gross Weight (lb)	Max. Dimension (1) (in.)	Drop Height (2) (in.)	Notes (6)
0 to 50	Under 36	22	(3)
50 to 100	36 to 48	16	(3)
100 to 150	48 to 60	14	(3)
150 to 200	48 to 60	12	(3)
200 to 250	60 to 66	27	(4),(5)
250 to 400	66 to 72	24	(4),(5)
400 to 600	72 to 80	21	(4),(5)
600 to 1000	80 to 95	18	(4),(5)

NOTES: (1) Any edge, diameter, or height.
(2) Applicable value is minimum indicated by weight and dimension criteria.
(3) Free-fall drop on all faces, edges, and corners
(4) Cornerwise rotational drop
(5) Also 5 miles/hour hard impact
(6) See Methods 5005, and 5023 in FED-STD-101C (Appendix B, Ref. 47).

TABLE 3-3: LIMIT LOAD FACTORS (G's) FOR TRANSPORTATION

Transport	X	Y	Z
Air (4)	+3.0 -1.5	+/- 1.5	+3.0 -2.0
Truck	+3.0 -1.5	+/- 0.5	+2.0 -1.0

NOTES: (1) For transportation loads, X is in the direction of travel and Z is along the local vertical, positive up.
(2) The loads are to be applied separately along each axis.
(3) These are input loads to the transportation mounts and are not the response loads of the module.
(4) These factors may change, dependent upon the transport aircraft utilized.

TABLE 3-4: AIRCRAFT TRANSPORTATION VIBRATION CRITERIA

JET AIRCRAFT INDUCED VIBRATION (5-2000-5 Hz @ 1 octave/min)

5 - 10 Hz	@ 0.599 mm (0.022 in.) Double Amplitude Displacement
10 - 35 Hz	@ 0.11 g peak
35 - 200 Hz	@ 0.043 mm (0.0017 in.) Double Amplitude Displacement
200 - 2000 Hz	@ 3.5 g peak

PROPELLER AIRCRAFT INDUCED VIBRATION (5-700-5 Hz @ 1 octave/min)

2 - 4 Hz	@ 10.67 mm (0.42 in.) Double Amplitude Displacement
4 - 5 Hz	@ 0.35 g peak
5 - 12 Hz	@ 0.35 g peak
12 - 55 Hz	@ 1.168 mm (0.046 in.) Double Amplitude Displacement
55 - 300 Hz	@ 7.0 g peak
300 - 700 Hz	@ 3.5 g peak

3.2 FLIGHT LOADS

(Per Planners Guide, Appendix C)
3.2.1 Liftoff & Ascent Loads.- The design environment imposed in association with the launch vehicle is presented in Appendix C, which is an extract of data provided in Ref. 53.

3.2.2 On-Orbit Loads.- Design limit loads and moments occurring on-orbit shall be determined for equipment operations where structural strength or stiffness is critical. Such loads and moments may be developed during separation of the spacecraft from the ascent vehicle, deployment of spacecraft appendages, operation of other moveable equipment/mechanisms while performing its mission, spacecraft maneuvering, and exposure to thermal gradients.

3.3 REENTRY AND LANDING LOADS (Not Applicable for MUADEE)

3.4 EQUIPMENT INSTALLATION LOAD FACTORS

(Per NASA/GSFC specified criteria)
The combined design limit load factor (applied in any direction) will be based on the mass of the instrument as illustrated in Figure 3-1 (see also Section 2.2.4).

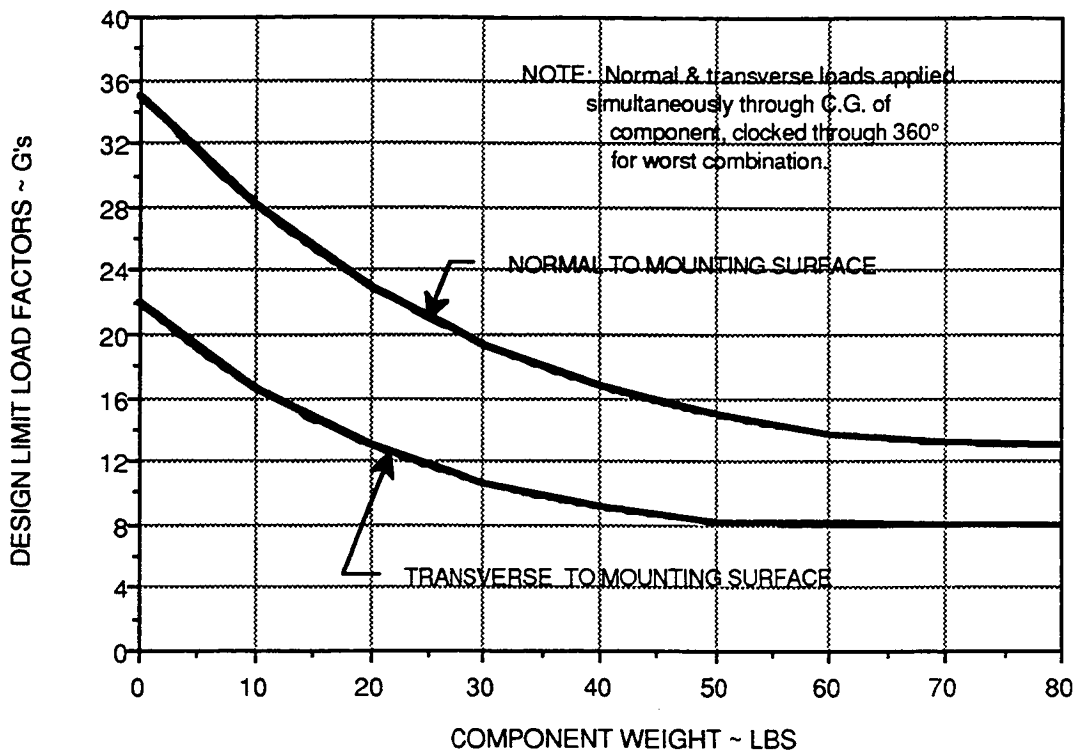

FIGURE 3-1: Equipment Installation Design Limit Load Factors

3.5 PRESSURES AND VENTING

Limit and design pressure-load criteria for active pressurized equipment/systems are included in the requirements for pressurized structures and equipment in Section 2.5. For both active and inactive enclosed structures/components ambient pressure changes are to be considered. Venting analyses may be required to determine limit pressures to be utilized in the design. Also, exposure to heat sources may affect the pressure criteria provided herein.

During ground transport a minimum range of ambient limit pressure variations is 86.1 to 109 kPa. However, depending upon the elevations along the route of travel, pressures as low as 6.89 kPa can be encountered.

During air transport a wide range of ambient pressures are potentially applicable, depending upon the altitudes to be flown by the transport aircraft and whether or not the spacecraft/payload is carried in a pressurized compartment or container. The carrier aircraft rates of ascent and descent affect the limit pressures to be encountered, and the requirement to consider aircraft operation emergencies (such as sudden decompression and/or rapid descents) is likely to impose major limit pressure conditions.

On-orbit limit pressure environments for some sealed assemblies can vary from a high of 109 kPa to a low of 1×10^{-10} torr. In the event that the temperature of the assembly is increased the high side value can go above 1.09 kPa.

3.6 OTHER ENVIRONMENTS

Structural design may be affected by environmental considerations not covered by the load, acceleration, pressure and thermal considerations defined above in Sections 2 and 3. The Spacecraft development effort shall address the following environments for possible applicability in defining critical limit conditions for design.

1. AMBIENT TEMPERATURES.-

The ground environments should be controlled as necessary to assure that they do not drive the design of space hardware. Typical ground condition design environments are:

Storage:	10° to 27° C.
Factory:	19° to 25° C.
Transportation:	15° to 27° C.
Launch Base:	-4° to 38° C.

The orbital environments for the equipment mounting interfaces are:

Normal:	0° to 30° C
Survival Modes:	-20° to +50° C

2. ENVIRONMENTAL HEAT TRANSFER

The spacecraft shall be designed consistent with the thermal flux parameters defined in Table 3.5.

TABLE 3-5: MUADEE Thermal Flux Parameters (Preliminary Data: To be verified)

	Minimum	Maximum
Solar Radiation	0.0129W/cm^2	0.142 W/cm^2
Albedo	0.275(ratio)	0.375 (ratio)
Earth Infrared Radiation	0.0233 W/cm^2	0.0222 W/cm^2

3. HUMIDITY.- Applicable ranges for humidity may vary from very dry to very moist (such as 5% to 100%), and/or to be controlled to within a very narrow range (such as 50% $\pm$ 10%).

4. OZONE AND ATOMIC OXYGEN.

The spacecraft shall meet performance requirements during exposure to atomic oxygen (AO) experienced during a (TBD) time period. Atomic oxygen fluence is shown in Figure 3-4.

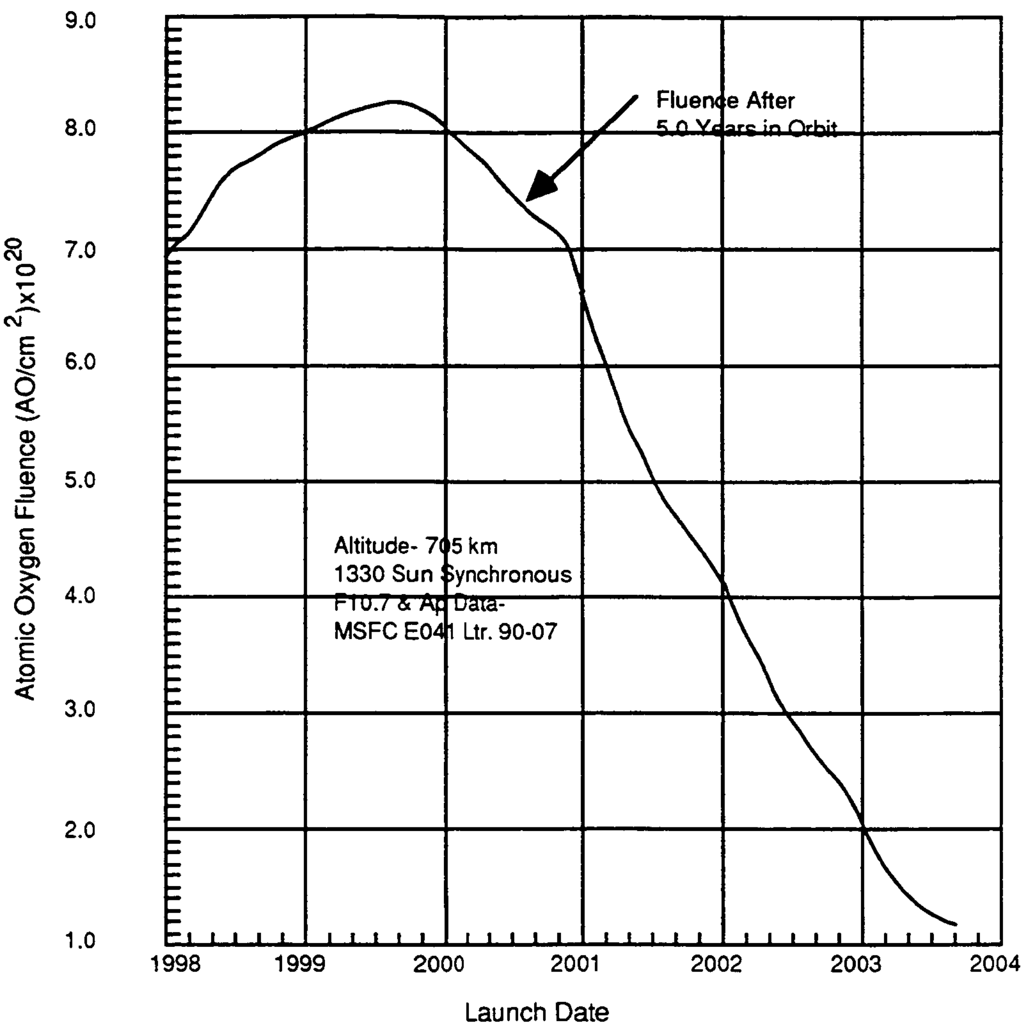

Figure 3-4 Atomic Oxygen Fluences for MUADEE (To be verified)

5. METEOROIDS AND SPACE DEBRIS.- Any applicable levels of exposure may or may not affect structural adequacy, and hence shall be defined. (TBD)

6. RADIATION.-The spacecraft shall meet performance requirements when exposed to the total dose due to the trapped proton, electron, and solar proton radiation environment which will be experienced by the Spacecraft as shown in Figure 3.5. The "Total Dose X Margin of 2" curve of Figure 3.5 shall be used for a design margin factor of 2. (To be verified).

The cosmic ray integral linear energy transfer (LET) distribution, as shown in Figure 3.6, shall be used for making single event upset (SEU) and single event latchup (SEL) calculations. Underl large solar flare conditions, the error/latchup rate will be approximately 1000 times higher. (To be verified).

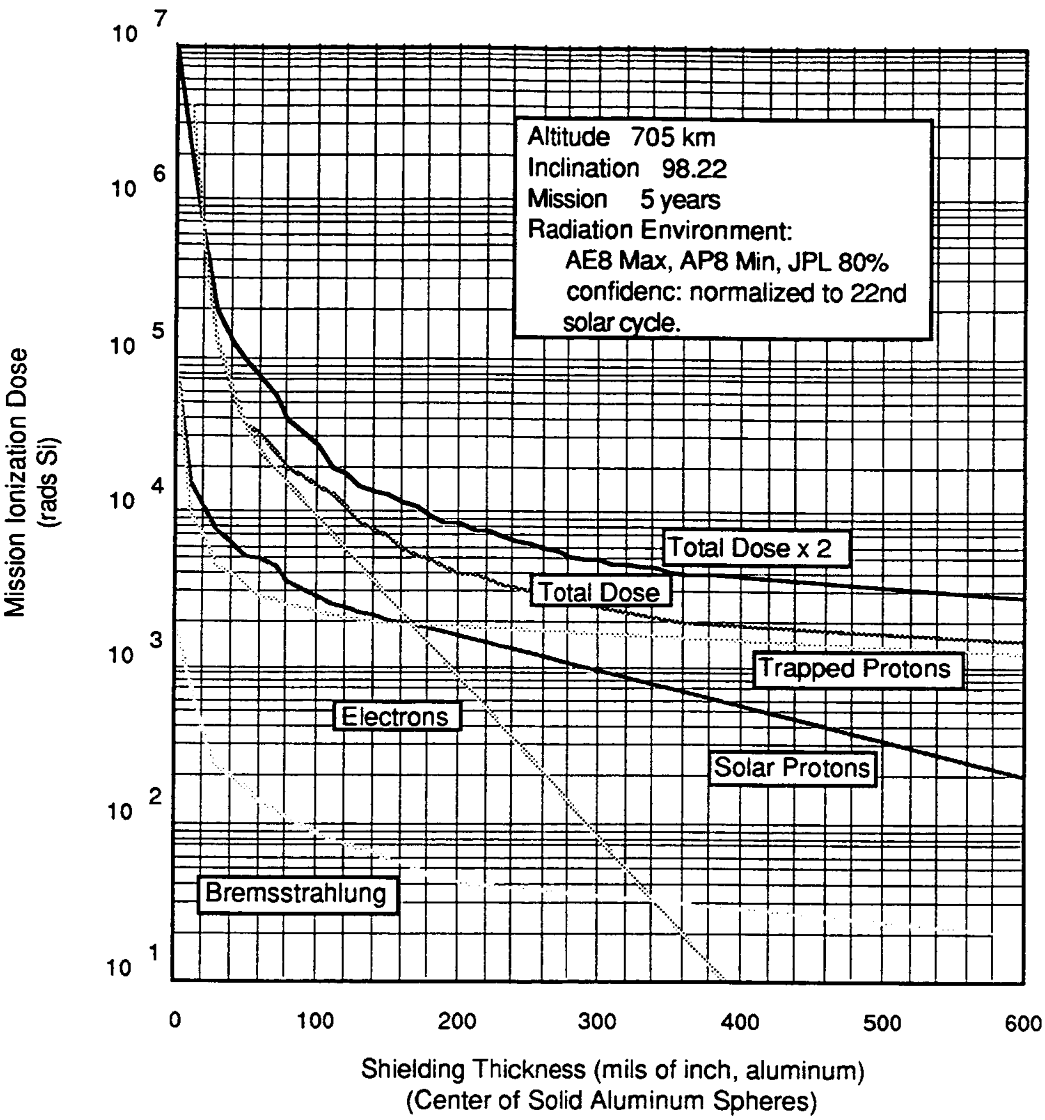

FIGURE 3.5: Total Ionizing Dose Radiation Environment

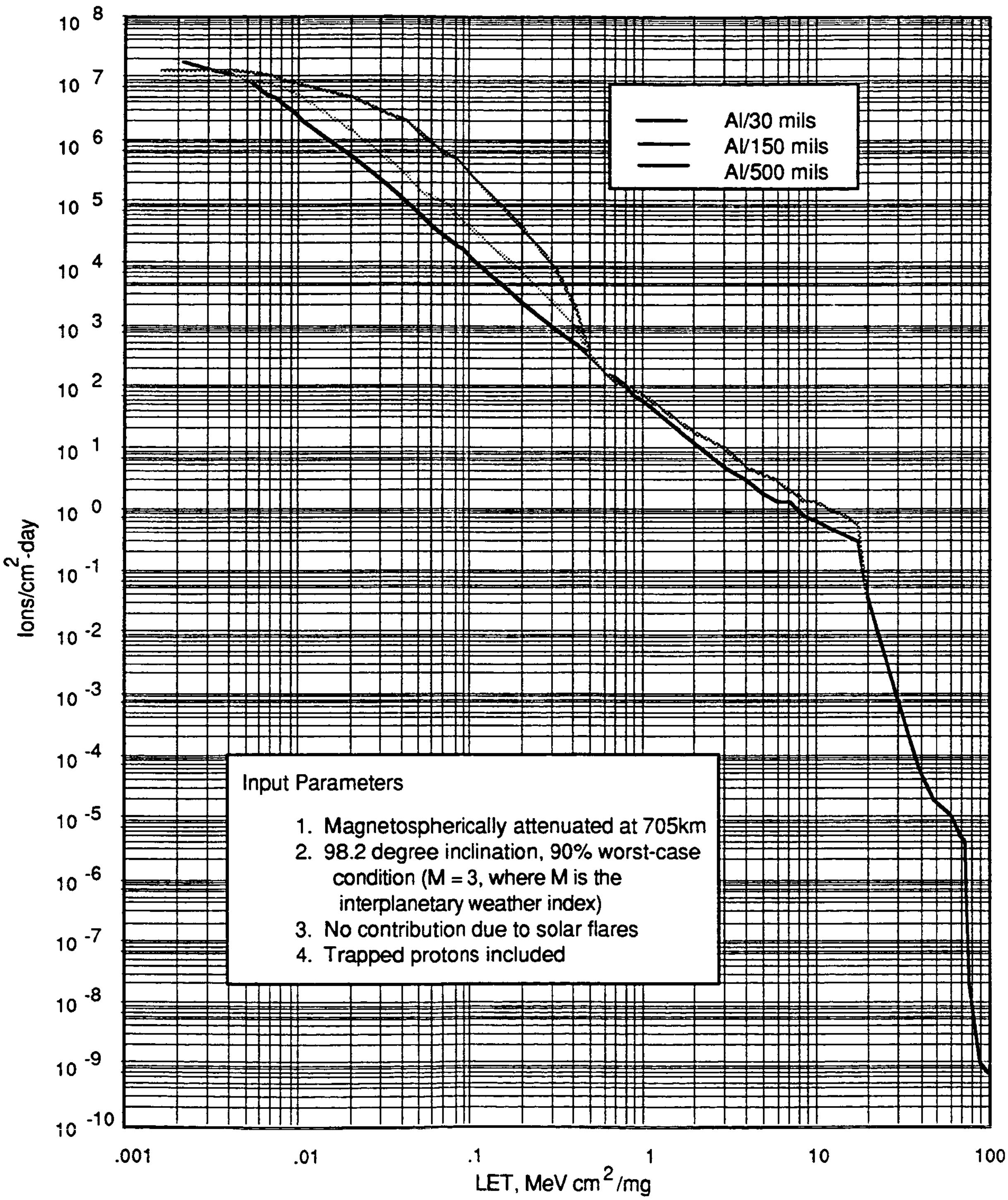

FIGURE 3.6: Cosmic Ray Linear Energy Transfer (LET) Spectrum

7. SHOCK & VIBRATION.- Acoustic imposed loading, and pyrotechnic device imposed shock loads, shall be considered in combination with other elements of limit loading, especially during liftoff and activation of any pyrotechnic devices. Specific environmental shock and vibration loads are presented in Appendix C. (TBD).

8. CONTAMINATION.- The instruments will be integrated with the Spacecraft in a Class 10,000 and test flow. The spacecraft design, including instrument layout, integration, test, ground handling, storage and transportation shall comply with MIL-STD-1246B Level 600A. Degradation of contamination sensitive hardware due to contamination on the ground and during all mission cleanroom environment and maintained in that environment as much as possible during all mission phases shall not be to such a degree as to prevent the hardware from meeting mission requirements during the 3-year (Earth years) on-orbit design life goal. (To be verified).

3.7 LIFE SPECTRUM LOADS

When fatigue or fracture control considerations are evident all load sources and environments shall be considered to determine the appropriate loading spectra for use in life analysis and the determination of fracture control procedures.

4.0 STRENGTH AND STIFFNESS REQUIREMENTS

Loading conditions for use in the design of structures are defined either by limit load factors or by limit axial, shear, torsion, and bending moment diagrams, and by use of a dynamics model with a load transformation matrix. Limit pressure distributions are to be included when applicable. Stiffness and other limitations, as described in this document, shall also be considered where appropriate. These, together with appropriate application of the Mass Properties (Section 2.1), the Design Factors (Section 2.4), the Thermal Analyses (Section 2.6), and the Material Properties (Section 2.7) provide the basic data required prior to performing the structural design/analyses. Specific criteria governing the strength analyses are provided hereunder.

4.1 DESIGN YIELD LOAD

The structure shall be designed to have adequate strength and stiffness to withstand design yield loads and pressures in the expected operating environments throughout its service life without experiencing excessive deflections (see Section 4.6) or detrimental permanent set. As defined in Appendix A, the design yield load is the product of limit load and the appropriate yield factor of safety (see Section 2.4.3). Combining loads to obtain a design yield load (or stress) is discussed in subsequent sections.

The allowable yield load, also defined in Appendix A, is a function of the elasticity of both the material and the structure. The allowable yield load is directly related to yield stress and strain. Classically, the yield stress is defined as the value at which a sharp break in the stress-strain curve occurs below the ultimate stress. At this point, the material elongates considerably with little or no increase in stress. Most non-ferrous metals and most high strength steels do not show this break, but yield more gradually to the point of rupture. Therefore, to limit permanent deformations of any appreciable amount, an arbitrary value of 0.002 inches/inch permanent strain was selected to establish a yield point on all stress-strain diagrams. The corresponding stress is called the yield stress in all standard military and government agency material documents. For practical purposes, this value may be determined for the stress-strain diagram by drawing a line parallel to the straight (or elastic) portion of the curve through a point representing zero stress and 0.002 strain.

In recent years, this arbitrary definition of allowable yield stress has been observed to be too conservative and in some cases not cost-effective for aerospace applications. Permitting strains greater than 0.002 is quite reasonable as long as the resulting deformation will not jeopardize the mission operation. Analyses and/or tests shall be performed to determine detrimental structural deformations. Caution shall be exercised in permitting a structural element to operate at or near yield for repeated load application (see Section 4.8).

4.2 DESIGN ULTIMATE LOAD

The structure shall be designed to have adequate strength to withstand design ultimate loads and pressures in the expected operating environments without experiencing rupture or collapse. As defined in Appendix A, the design ultimate load is the product of limit load and the appropriate ultimate factor of safety. Combining loads to obtain an ultimate load (or stress) is discussed in subsequent sections.

4.3 COMBINED LOADS

Loads resulting from different sources which occur simultaneously, including preloads, shall be considered to be applied singly and in rational combinations to result in the maximum loads on structural elements. Acoustic induced loads are to be combined with liftoff and/or other in-flight loads unless it can be established that these loads do not contribute significantly to the combined loads. When internal pressure acts to stabilize the structure while it is simultaneously subjected to compression loading, minimum operating pressure (accounting for valve tolerances) shall be used. The ultimate or yield factor of safety shall not be applied to any relieving load.

Thermally induced strains and loads shall be combined in the analysis, without additional factors of safety, to determine the critical design load (limit, yield, ultimate, etc.). The manner in which this combination is made shall consider the strain cycle as applied in service.

4.4 TEMPERATURE EFFECTS

The structure shall withstand, without failure or excessive deformations, the effects of thermal-energy transfer (due to natural, man-made, and induced thermal environments) on structural and thermal protection systems (including insulation materials) material properties. The effects of temperature and temperature gradient (including temperature uncertainties defined in MIL-STD-1540 (Appendix B, Ref. 23)) shall be considered in the analysis without an additional factor of safety. External loads and temperature-induced loads (e.g., tank pressure increases due to propellant residuals and heating) must account for interacting effects whenever the conditions of plasticity or creep are present.

4.5 STRUCTURAL INSTABILITY

All structural components that are subject to compression and/or shear in-plane stresses under any combination of ground or flight loads including loads resulting from temperature changes, shall be investigated for all modes of structural instability failure (from general buckling to local crippling). Design loads for buckling shall be treated as ultimate loads. Any load component that tends to alleviate buckling shall not be increased by the ultimate factor of safety. External pressure or torsional (destabilizing) limit loads shall be increased by the ultimate factor of safety, while internal pressure (stabilizing) limit loads shall not be increased (see Section 4.3).

Controlled buckling is a load state in which portions of the structure are designed to function in the buckled mode; e.g., tension field shear beams and the initial panel buckling between longerons. Controlled buckling shall be permitted with the proviso that the stiffness or deformation requirements are not compromised.

4.6 STIFFNESS DESIGN

4.6.1 Limit Load Conditions.- Elastic or inelastic displacements at design limit loads shall produce no adverse effects to the success of the mission, operation of mechanisms, or performance. Structural displacements, deflections, or deformations are considered excessive and detrimental if, after allowance for thermal deformations:

1. Unintentional contact, misalignment or divergence between adjacent components occurs.
2. A component exceeds the dynamic space envelope established for that component.
3. The strength or rated life of the structure is reduced below specified levels.
4. The effectiveness of thermal protection coatings or shields is degraded.
5. The proper functioning of components is jeopardized.
6. Personnel safety is endangered.
7. The functional characteristics of the vehicle are degraded below specified limits.

8. Confidence is reduced below acceptable levels in the ability to ensure flight-worthiness by use of established analytical or test techniques.

9. Leakage above specified rates is induced.

10. The material changes its characteristics beyond specified limits.

11. Excessive dynamic response amplitudes or load factors occur or adverse dynamic coupling is induced.

12. Misalignments of components exceed allocated budgets

Rigorous analyses shall be performed to insure that adequate stiffness is provided so that displacements (deformations) occurring between limit and ultimate loads do not precipitate an associated failure mode in other structural or functional components. Accumulated deformations shall be anticipated and accounted for in the design. Loading and temperature-induced deformations that may induce loading across interfaces or restricted clearances shall be determined and considered in the design. Dynamic-response effects of mated vehicle components shall be determined and accounted for in the design. Joint slippage and deformation shall be considered.

4.6.2 Ultimate Load Conditions.- Adequate strength shall be provided so that structural deformations shall not precipitate structural failure during any design conditions and environments at loads less than ultimate load.

4.6.3 Dynamic Properties.- Dynamic stiffness characteristics for systems, subsystems, and/or components are stated in terms of frequency ranges for specified types (e.g., axial, bending, etc.) of structural natural vibrational modes. Requirements on specific dynamic stiffness properties are based on minimizing coupling interaction with known higher assembly characteristics, isolating or avoiding frequencies where known excitations are present, or minimizing analysis complexity.

4.6.4 Component Stiffness.- Component stiffness may be specified in terms of vibrational frequency to control dynamic interactions, loads, vibrations, control systems, etc., or in terms of force/deflection or moment/angle to satisfy deflection, clearance, static stability, or higher assembly dynamic characteristics.

4.7 MARGINS OF SAFETY

The relative strength or stiffness capability of structural components or assemblies shall be evaluated for all critical design conditions. The structural design margin of safety (see Appendix A) shall be based on theoretical analysis, and substantiated by test whenever possible. The margins so determined shall be used as final indicators of available strength, or adequate stiffness, after all other design characteristics, conditions, and factors have been accounted for. The margin of safety shall be positive, and shall be determined at design ultimate and yield levels. For minimum-weight design, the margins of safety should be as small as is practicable, ideally zero.

As an alternative approach to showing the available strength or adequate stiffness using tables of margins of safety, tables of calculated allowable yield or ultimate factors of safety may be used as final indicators of the available strength or adequate stiffness.

4.8 SERVICE LIFE

(Per U. of M System Requirements)
The spacecraft shall be designed for a minimum on-orbit lifetime of three Earth years after launch.

4.8.1 Fracture Control.- Aerospace vehicle structures, be they primary, secondary or pressure vessels, are vulnerable to the initiation and propagation of cracks or crack-like defects during their service life, which may lead to structural failure. The term "fracture control" is used to describe the approach to design, manufacture, and quality control which seeks to prevent structural failure due to cracks or crack-like defects. Fracture control criteria are applicable to the vehicle /payload and/or its systems, assemblies, or components which are determined by engineering analysis and/or tests to be susceptible to cracking or fracture on the basis of anticipated loads and environment, and which are critical to safety or system performance. These criteria are not intended to apply to accidental or inadvertent mishandling which in itself would cause failure.

The fracture control criteria define measures covering the entire operational life of the vehicle, including engineering design, material selection and procurement, fabrication processes, quality assurance procedures, acceptance and/or periodic proof tests, flight tests, and operational service usage. Fracture control measures also apply to non-flight articles undergoing development and qualification tests.

4.8.2 Fatigue & Creep Strength.- Fatigue and creep strength shall be provided to satisfactorily withstand the service life spectrum of loads and environments specified in Section 3.7 with the life exposure safety margin specified in Section 2.4.11. Regardless of whether fracture control or conventional safe-life analysis procedures are used, in order to avoid metal fatigue problems and achieve a fatigue resistant structure there are two important principles to follow: (1) design the structure to be fail-safe, and (2) keep stress concentrations as low as possible and thereby avoid the weight penalty caused by decreasing the stress to obtain the desired fatigue life. To minimize stress concentrations utilize: (1) close, or interference fit fasteners, (2) excess connectors to reduce bearing, (3) shot peening, and (4) designs that minimize joint eccentricity.

4.8.3 Creep.- Materials which are resistant to creep shall be used. Structural components shall not exhibit cumulative creep-strain leading to rupture, excessive deformation, or creep buckling during their service lives. For creep-critical structural components, analyses should be supplemented by tests to verify the creep characteristics for the critical combinations of loads and temperatures for specified times and environments.

4.8.4 Corrosion.- Corrosion prevention procedures shall be initiated during the design of the vehicle/payload. There are several types of corrosion: galvanic, stress, fretting, and intergranular corrosion. The following paragraphs present the methods of prevention for each.

1. GALVANIC CORROSION.- Galvanic Corrosion can result from electro-chemical reaction of two dissimilar metals in physical contact (see Appendix B, Ref. 27). To prevent and/or control galvanic corrosion, the following procedures are recommended:

    a. Select metal combinations close together in the galvanic series.
    b. Use large anodic (sacrificial) metal areas in combination with small cathodic metal areas.
    c. Review proposed material combinations and platings for compatibility.
    d. Avoid the use of dissimilar metal combinations in corrosive environments.
    e. Insulate dissimilar metals either by protective coatings or by interposing an inert material between them.
    f. Eliminate access of water to metals by sealants, protective coatings, etc.

2. STRESS CORROSION.- The combined effect of a susceptible material, a corrosive agent, and stress can result in a premature structural failure. The process can be accelerated by exposure to elevated temperatures. The phenomenon is called Stress Corrosion cracking. Criteria for controlling stress corrosion are provided in MSFC-SPEC-522 (Appendix B, Ref. 26).

3. FRETTING CORROSION.- Corrosion can occur at the interface of two highly loaded members where relative motion between these members occurs. The combined action of mechanical wear and chemical corrosion can result in Fretting Corrosion. To prevent and/or control this action, the following procedures are recommended:

a. Use a lubricant on the contact surface.
b. Design the joint with a hard metal in contact with a soft metal.
c. Seal the joint to prevent entrance of a corrosive agent.
d. Consider adhesive bonding in addition to mechanical attachments for fracture-critical parts.

4. INTERGRANULAR CORROSION.- Intergranular Corrosion is cause by potential gradients between grains and adjacent grain boundary area. It is most aggressive on exposed short transverse grain. This condition exists at the end of plate, bar, and extrusions; at the parting plane of forgings; and at lap joints, seams and fastener holes. Exfoliation is a severely destructive form of intergranular corrosion characterized by the actual leafing out of corroded section of metal away from the rest of the part. To prevent and/or control this action, the following procedures are recommended:

   a. Avoid exposure of short transverse grain structure.
   b. Use a protective film such as plating (Cadmium is prohibited for space applications), cladding, anodizing, or a sealing compound.
   c. Use a cold working process on the surface grain structure such as shot peening.
   d. Use alloys and heat treat conditions least susceptible to intergranular corrosion.

4.9 SUPPLEMENTARY REQUIREMENTS

4.9.1 Structural Non-linearities.- Structural non-linearities occur from three causes: unknown irregularities in the structural assembly, inelasticity, and geometrical non-linearities. These nonlinearities impede the ability to adequately predict loads and response and should therefore be minimized.

1. UNKNOWN IRREGULARITIES.- This source can affect the vehicle response of the structure at all frequency levels, and therefore cannot be ignored. The predominant source for this problem is joint slippage. Critical structural joints should be evaluated carefully, and (possibly) tested, to minimize the problem. Careful manufacturing control can alleviate the problem. Possible solutions for critical joints are to use metal bonding and/or close-tolerance mechanical fasteners.

2. INELASTICITY.- This source occurs when the vehicle /payload significantly exceeds the allowable yield strength of elements within the structure. Careful structural analysis shall be performed to insure that local inelasticities will not adversely affect the performance of the assembly.

3. GEOMETRIC CHANGES.- This source occurs when the vehicle /payload sustains large deflections not accounted for in the conventional small-deflection theory of analysis. A common example is the beam-column. Another structural cause is the pretensioning consideration occurring from membrane action on a thin plate subjected to lateral pressure. Such occurrences must be considered if they are suspect.

4.9.2 Design Thickness.- The structural design thickness, t(d), for each structural member other than pressure vessels shall be the minimum of the thickness calculated by both of the following methods:

t(d) = Mean thickness based on equal plus and minus tolerances
t(d) = N times the minimum thickness, N = 1.10 for strength critical designs
N = 1.05 for buckling critical designs

The mean and minimum material thicknesses, used above, shall account for cumulative damage to the physical properties of the material resulting from repeated exposure to the specified operating environment. The design thickness for pressure vessels shall be the minimum thickness.

4.9.3 Dimensional Tolerances And Misalignments.- Design and "as fabricated" dimensional tolerances shall be accounted for in all structural analyses. The effects of allowable structural misalignments, control misalignments, and other permissible and expected dimensional tolerances shall be considered in analysis of loads, load distributions, and structural adequacy.

4.9.4 Preloads.- Structural interfaces shall be preloaded whenever possible to minimize non-linear response to cyclic loads. (See also Section 4.9.8-6).

4.9.5 Composite Structure.- The use of filamentary composites as a structural material permits the design engineer to tailor structures to specific requirements: strength, stiffness, and/or thermal stability. Unlike most metallic structural materials, the properties of composites are anisotropic, but may be effectively quasi-isotropic. The properties of a laminate depend not only on the chemical composition of the constituents but also on their geometry and orientation. Because of the complexity of the many variables in the design and analysis of composite structure, computerized procedures must be used for their evaluation. Several composite analysis codes are available at LMSC which provide capability for:

1. Design and/or analysis of mechanical stiffness, strength of anisotropic composite laminates, buckling of plates, cylinders, and columns, and thermal properties.

2. Elastic analysis of single/double lapped joints.

3. Design and/or analysis of mechanical stiffness/strength of anisotropic composite laminates, perform in-plane and out-of-plane analysis of laminate plates subjected to normal and in-plane loads and shears. Computation of section properties of complex beam composite laminates.

4. Elastic/plastic, linear/non-linear time-dependent (creep) analysis of laminates subjected to in-plane axial and out-of-plane bending loads; stress or strain conditions can be used. Includes moisture and temperature effects, and metal matrix composite behavior.

(Per NASA/GSFC requirements)
Currently there are no acceptable methods for applying fracture control technology to composite structures. A material design factor is required (see Section 2.4.7) and composite elements must be tested (see Section 2.4.3 and 3.4.7) because of the observed discrepancy between allowable strength and other material properties data based on material specification values for laminates and the observed strength of production parts. Material properties, design standards and specifications are presented in the LMSC Design Handbook. Refer also to special structural bulletins.

4.9.6 Honeycomb Structure.- Bonded honeycomb sandwich construction has been a basic structural technique in the aerospace industry for many years. Virtually every aerospace vehicle flying in the atmosphere or in space depends, in part, upon the integrity and reliability offered by this type of structure. The design and fabrication is governed by the criteria and specifications presented in MIL-HDBK-23 (Appendix B, Ref. 15). Special considerations affecting design criteria relate to: (1) perforated or non-perforated core (affects pressure loads and potential for contamination during outgassing); and (2) attachment modes between the honeycomb and other structures.

Procedures shall be devised for reliable detection of flaws in bonded or brazed construction. Non-destructive testing of all bonded and brazed sandwich structure shall be performed, and consideration should be given to performing tests to verify attachment designs.

4.9.7 Welds.- A weld is the localized coalescence of metal wherein coalescence is produced by heating to suitable temperatures, with or without the application of pressure and with or without the use of filler metal. There are a number of types of welding used: fusion, electron beam, flash, braze, etc. Welding primary structure is limited to two methods: fusion and electron beam welding.

Welds shall be classified for structural applications as follows, as a means for designating levels of inspection and other special requirements. The appropriate class shall be designated by Stress personnel in accordance with this document and incorporated in a drawing note by Design personnel.

1. CLASS I: VITAL WELDS.- Welds that could cause catastrophic failure of the spacecraft or one of its major components, or cause injury to personnel, if a single failure of such a weld would occur during an operating condition. Class I welds are further classified as follows:

   Class I (A).- Welds that are not fail-safe.

   Class I (B).- Welds that can be shown by analysis or test to be fail-safe. Class I (B) welds require the same type of inspection as Class I (A) welds.

2. CLASS II: NON VITAL WELDS.- This class includes all welds not falling within the vital classifications. Class II welds are further classified as follows:

   Class II (A).- Welds for which a high level of confidence is required.

   Class II (B).- Welds that are of secondary importance, non-structural, with minimal service requirements.

3. MATERIALS, DESIGN AND PROCEDURES.- Weldable materials, dissimilar material combinations, procedure specifications, and design aids are presented in the LMSC Design Handbook (Appendix B, Ref. 8).

4.9.8 Fasteners.- The 160 KSI fastening system is a family of CRES (A-286) or Titanium (6AL-4V) close tolerance bolts and companion CRES (A-286) platenuts which has been developed to eliminate the following problems: galled threads, broken platenut domes, cammed out driver recesses, requirement to shim platenuts, and requirement to stock various bolt-thread lengths. In the following paragraphs limitations are prescribed for the use of fasteners in spacecraft structures.

1. BOLTS.- The minimum size bolt to be used in primary structure is 1/4-inch diameter, except for cases where failure of a single bolt will not cause failure of the structure (such as multiple bolt patterns, seam bolting, etc.). In structural applications designated as "Fail-Safe," No. 10 fasteners may be permitted with special approval by the Project Stress Group Engineer. Fasteners smaller than No. 10 shall not be permitted for use in primary structural applications. The responsibility for determination of the application as "primary" or "secondary" rests with the Project Responsible Stress Engineer and the acceptance of this responsibility is signified when the drawing is signed.

2. NUTS.- Nuts designed to develop the ultimate tensile strength of the bolts are required in applications which utilize the allowable bolt loads. Special nuts which depend on friction for their anchorage and rigidity, such as clinch nuts, spline nuts, single rivet plate nuts, etc., are not acceptable for use in structural applications.

3. WASHERS.- High strength steel washers (LMSC-LS8999) shall be used under the nut and under the bolt head of 160 KSI and stronger bolts which are subjected to high tension loads. Counter-sunk washers must be used under bolt heads where necessary to provide clearance for the fillet. Aluminum washers may be used in shear load applications as a weight savings measure.

4. DESIGN ALLOWABLE STRENGTH.- The design allowable tension strengths for the 160 KSI fasteners system are prescribed in MIL-HDBK-5F Section 8.1.5, Table 8.1.5 b(2) (Appendix B, Ref. 13). These strengths are for rolled threaded fasteners. The corresponding design allowable shear strengths for the 160 KSI fasteners system are prescribed in Table 8.1.5 (a), using the unit shear stress of 95 KSI.

5. COMBINED LOADING.- In lieu of the interaction of combining tension with shear load prescribed in MIL-HDBK-5F, it is the policy in LMSC to use the following interaction equation:

$$M.S. = \frac{1}{\sqrt{R_T^2 + R_S^2}} - 1$$

where: M. S. = Margin of Safety

$$R_T = \frac{\text{Applied ultimate tension load on fastener}}{\text{Allowable ultimate tension load on fastener}}$$

$$R_S = \frac{\text{Applied ultimate shear load on fastener}}{\text{Allowable ultimate shear load on fastener}}$$

6. BOLT TENSION- The effects of preload shall be included when determining the tension load in the bolt. The ultimate and yield safety factors are applied to the external joint limit load when the ultimate and yield bolt tension loads are calculated. The ultimate tension load in the bolt shall not exceed the ultimate strength of the bolt. The yield tension load on the bolt shall not produce a stress, based on the thread root area, greater than the yield stress of the bolt material. Bolts subjected to sustained tension loads, including preload, shall be limited to 60 percent of the allowable ultimate tensile strength of the bolt. This requirement is to preclude exceeding K_{TH}.

7. CORROSION.- The fastener system shall be designed to prevent corrosion in accordance with the guidelines given in MIL-STD-889 (Appendix B, Ref. 27). The use of cadmium on any fastener system component is specifically prohibited.

4.9.9 Meteoroid Protection.- Meteoroids and space debris are solid particles moving in inter-planetary space and originate respectively from cometary and asteroidal sources, and from components, assemblies, or debris from manmade vehicles/payloads. Damage to vehicles operating in space can occur from the impacts of meteoroids and space debris because of their velocity, density and mass. The type and extent of the damage depends upon vehicle size, vehicle structural configuration and exposure time in space, as well as on meteoroid/ or debris characteristics. Such impacts on a spacecraft can result in damage such as: a puncture of a heat pipe or propellant tank; the deterioration of optical windows, protective surfaces and solar arrays; and degradation of thermal coatings by cratering or spalling. Other possible impact effects include damage to antenna systems, thruster nozzles, and electrical leads.

Meteoroid and space debris fluxes and densities have been determined from earth measurements, using photographic and radar techniques and from satellites and rockets using data from penetrations of membranes. These data have been compiled with considerations given for limitations of each method of observation, and models have been developed for design purposes. The vehicle/payload shall be designed, in accordance with guidelines prescribed by the Procuring Agency, to prevent meteoroid damage to structure or components which could impair flight worthiness or reduce the service life within a prescribed probability.

4.9.10 Radiation Protection.- The inner and outer Van Allen radiation belts, plus other sources of energetic proton and electron atomic particles, can cause radiation damage through two mechanisms: ionization and atomic displacement. Ionization is the removal of electrons from the atoms of the impacted materials, and is the chief damage mechanism to plastics, elastomers, oils and greases, glasses and ceramics. Radiation damage to metals is primarily due to atomic displacement. Displacement is also important in inorganic insulators such as glass and ceramics.

The need for radiation shielding shall be assessed, and when specified by the Procuring Agency, shall be provided as necessary to prevent allowable radiation doses and dose rates from being exceeded for the duration of the mission. When provided, shielding shall be compatible with the combined radiation, thermal and mechanical environments.

5.0 DESIGN VERIFICATION ANALYSES

(Per NASA/GSFC specifications)
The NASA Structural Analysis (NASTRAN) program shall be used for analyzing the structural design. The Thermal Radiation Analyzer System (TRASYS) shall be used for thermal geometry models and the Systems Improved Numerical Differencing Analyzer (SINDA85) or later version shall be used for thermal analysis.

5.1 MASS PROPERTIES ANALYSIS

Mass properties analyses shall be performed in accordance with the requirements delineated in Section 2.1 to provide the mass property data essential to the determination of vehicle dynamic response characteristics, limit loads, and structural shear, bending moment and torque design conditions. These analyses shall be updated periodically as required to assure that the structural design conditions continue to be valid (or conservative).

5.2 EXTERNAL LOADS ANALYSIS

External loads analysis shall be performed in compliance with the criteria delineated in Section 2.2 to determine the loads and environments expected to be imposed on the vehicle during its service life. All critical loads and load combinations shall be defined. Analyses shall account for vehicle geometry, flight conditions and environments, mass distributions, vehicle vibration-modal characteristics and structural damping, structural interaction with the control systems, variation of loads with time for deterministic load analyses, and all statistical loads for probabilistic load analyses. Computed static and dynamic loads shall be combined with thermal and other applicable environmental effects, to produce vehicle critical design loads, vehicle test loads, and data for use in establishing strength and operating restrictions on the vehicle. In the early design stages, quasi-static limit loads may be formulated using available data and appropriate uncertainty factors to enable detail design to proceed.

5.3 INTERNAL LOADS & STRESS ANALYSES

The internal loads and stress analyses shall be performed to verify structural adequacy in compliance with the criteria presented in Section 4.0. The stress analyses shall cover the structural responses to the critical loads, environments, and temperatures anticipated during the service life of the vehicle. These analyses shall define the critical combination of vehicle configurations, loads, environmental conditions, material properties, and interactions which determine stress levels and margins of safety for all structural components. Analyses shall also be performed to show that deformations do not cause degradation of the vehicle performance, or violation of allowable space envelopes, at limit load conditions. Stress analyses shall also provide data for use in establishing vehicle strength and operating restrictions.

5.4 STRUCTURAL DYNAMICS ANALYSES

Structural dynamic response and stability analysis shall be performed in compliance with the structural dynamics criteria delineated in Sections 3.1, 3.2, and 3.3. These analyses shall account for the vehicle geometry, ground handling and flight conditions, mass properties, vehicle stiffness distribution, and control system structural interactions and stability margins. Where applicable, the acoustic environment shall also be included. Where adequate theoretical analysis does not exist, or where experimental correlation with theory is inadequate, the analyses shall be supplemented by tests.

Dynamics analyses shall also be performed for vehicle systems where movements of mechanical devices and/or thermal/structural displacements of deployables or other subassemblies create dynamic movements of the vehicle (or components) which are critical to on-orbit performance.

5.5 VENTING ANALYSES

(Per Spec, Appendix C)
Venting analyses shall be performed, when pressures affecting limit load conditions are likely to be encountered, to verify the adequacy of the venting provisions (per Section 2.9) and to determine the pressure contributions to limit conditions in compliance with Section 3.5. The launch vehicle static pressure profile is provided in (TBD) of Appendix C.

5.6 THERMAL ANALYSES

Thermal analyses shall be performed in compliance with the criteria delineated in Section 2.6. Steady-state and transient thermal analyses shall be performed to account for conditions which affect heating, and in turn affect structural materials and their properties, structural components and their assembly, and thermal control materials. These analyses shall define temperature extremes of structural elements, design temperatures of components, and worst case temperature gradients for use in the thermal distortion analyses.

5.7 FRACTURE CONTROL ANALYSIS

Fracture mechanics analyses shall be performed , in accordance with the requirements of Section 4.8.1, when the service life requirements of the mission indicate the advisability of exercising fracture control procedures. Early agreement should be obtained with the Procuring Agency with respect to both the need for such analyses, and the detail definition of the scope of fracture control to be implemented.

5.8 DEPLOYMENT & SEPARATION ANALYSES

Analyses of mechanical deployment motions, loads and clearances between the vehicle, deployables, fairings, cargo bays, containers, separation structures and/or major subassemblies shall be performed with respect to the dynamic response characteristics determined in compliance with the requirements of Section 2.2.2. Mechanisms analyses shall include deployment rate, time, release and lock-up loads, and force or torque margin at environmental extremes. Also, where applicable, mechanisms analyses shall include preload requirements, lubrication adequacy, bearing loads and life prediction, and motor stability and performance. Separation analyses shall include worst-case predictions of clearances, separation rate, and tip-off rates. Appropriate design safety factors shall be applied to analysis-determined values to assure that the analysis and physical uncertainties involved in each design configuration have been adequately covered.

5.9 METEOROID & SPACE DEBRIS PROTECTION ANALYSES

Meteoroid and space debris vulnerability/protection analyses shall be performed, in accordance with Section 4.9.9, when such environments are critical to the structural design for accomplishment of the mission. Early agreement should be obtained with the University of Michigan with respect to both the need for such analyses, and the detailed definition of the environmental data to be utilized.

6.0 DESIGN VERIFICATION TESTS

Tests shall be conducted to: (1) provide developmental engineering data to augment design/analysis, manufacturing process control and inspection tasks; (2) verify the weights of equipment, assemblies, and the entire vehicle/payloads; (3) verify thermal, structural dynamic, and stress analytical math models; (4) demonstrate structural integrity, identify modes of failure and determine margins of safety; and (5) insure that the structure and mechanical systems are functionally satisfactory and compatible with other systems when operated in simulated operating environments. As a guide in planning test programs, reference should be made to MIL-STD-810, MIL-STD-1540, and MIL-HDBK-340 (Appendix B, Refs. 22, 23 and 46), and early agreement should be obtained with the University of Michigan with respect to both the need for major tests and the detail definition of the test programs. Tests, in general, either augment or verify design/analyses. Where test conditions, such as is often true in the case of random vibration testing, may not be preceded by design/analyses, care shall be exercised in the selection of test levels and in the conduct of tests on flight hardware. In no case shall test levels in excess of previously established design environments be imposed unless associated risks have been evaluated and judged acceptable.

6.1 TYPES OF TESTS

6.1.1 Developmental Tests.- Developmental tests shall be conducted to provide engineering design and test information to validate analytic techniques and assumed design parameters, uncover unexpected system response characteristics, evaluate optional designs, and evaluate test procedures. These tests may be accomplished on simulated components, assemblies or the entire vehicle in early stages of design concept selection or detail design.

6.1.2 Qualification Tests.- Qualification tests are performed to provide verification that a component, assembly, subsystem or an entire vehicle has met its full performance/design requirement(s). Test levels and durations normally exceed the expected flight limit levels to ensure that the specified design safety margins (or at least some portion of them) are available. Test levels shall exceed the acceptance test (see Section 6.1.4) levels. On completion of a successful qualification test the unit is considered no longer usable on the flight vehicle, except in special circumstances where refurbishment of the unit is undertaken. In the case of a Structural Test Vehicle (STV) (see Appendix A), some of the tests are carried out to levels beyond design ultimate load and to actual structural failure to demonstrate the test ultimate margin of safety that is available above design ultimate loads.

6.1.3 Protoqualification Tests.- As defined in Appendix A, tests of a flight vehicle to environments more severe than acceptance tests but less severe than qualification tests are identified as protoqualification tests. This mode of test verification is employed when a qualification test article cannot be made available, but nonetheless some level of test verification above acceptance tests is considered necessary. Thorough inspection for possible damage is essential on completion of the tests, and some refurbishment may be necessary prior to certification of flight readiness.

6.1.4 Acceptance Tests.- Acceptance tests are performed primarily to detect latent material and workmanship defects and to verify that adverse tolerance buildup does not result in "out of specification" performance. Test levels and duration on lower-level assemblies shall generally be equal to or exceed those imposed on higher assembly levels. Acceptance test levels shall be equal to or greater than the limit levels predicted for flight, to verify that the assembly can perform properly at the maximum level expected to be encountered during its actual mission.

6.2 MASS PROPERTIES VERIFICATION

Flight article final mass properties shall be verified to provide affirmed properties for use in operational planning, and to assure that structural design mass property limits are not exceeded. Verification may be accomplished by analysis, test measurement or a combination of both techniques. In general, weight and center of gravity shall be verified by test measurements. Inertial properties shall normally be confirmed by analysis, except for dynamically balanced spin-stabilized spacecraft which shall normally be verified by test. Verification measurements shall be performed when the vehicle is in essentially a complete configuration, preferably during systems test, and is as free of tare weights as is practicable.

6.3 STRUCTURAL STRENGTH & STIFFNESS TESTS

6.3.1 Static Tests.- The ability of the structure to sustain all critical design loads and environmental conditions in the manner required may be demonstrated by structural static tests. Such tests shall be performed on full scale representative specimens of individual components, or assemblies of components, as agreed upon with the University of Michigan. When environmental conditions cannot be properly simulated in these tests, allowances for material properties, combined loadings and other missing effects shall be provided in the test configuration and loads. Where prior loading histories affect the structural adequacy of a test article, appropriate simulation or allowance for these shall be included in the test plans. Adequate instrumentation shall be provided in order to properly evaluate the relationship between load levels, displacements, stresses, yielding, and/or ultimate failure as is appropriate for the conditions and structures being tested. Structural tests that have previously been performed for other programs may be substituted for tests required by a new vehicle/payload, provided that the prior and new structural components are comparable and the applied loading is similar.

These tests shall verify that the general structure does not experience detrimental yield at design yield loads, and/or detrimental deformations at limit loads and pressures, in accordance with the criteria of Sections 4.1 and 4.6. The tests shall further verify that the structure does not rupture or collapse at ultimate load and pressure in accordance with the criteria of Sections 4.2 and 4.5. Static tests may also be performed to verify structural stiffness characteristics (structural compliances) utilized in the dynamics analyses.

For structure classified as pressure vessels, at least one specimen typical of flight hardware shall be tested to demonstrate that each vessel is capable of sustaining ultimate pressure without rupturing (bursting) in accordance with Sections 2.5.2 and 4.2. Each test specimen shall be of the same design as planned for the flight hardware, and shall be fabricated from the same materials and by the same process specifications planned for production of flight hardware. The effects of operating temperatures and environments shall be simulated as appropriate in the tests, generally by introducing equivalent load increments above limit loads.

6.3.2 Fatigue & Creep Tests.- Fatigue and/or creep tests shall be performed, as required, on representative components to demonstrate adequate fatigue and/or creep life after being subjected to four times a representative flight spectrum of loads and/or pressures.

6.4 STRUCTURAL DYNAMICS TESTS

6.4.1 Modal Surveys.- Modal tests shall be performed to confirm the analyses specified in Section 5.4. Modal tests are typically performed on the assembled satellite system, but may also be desirable on major subsystems. The modal characteristics required to be determined are established under consideration of the frequency and spacial characteristics of the critical forcing functions. The characteristics to be determined from the test are mode shapes, frequencies and modal damping for all structurally significant modes. Normally for the satellite system, main structural loads and responses derive from modes with frequencies less than 50 Hz. Typically, 50 Hz is the upper bound on the frequencies of modes to be determined, but in some cases can be lower when justifiable. Mode shapes exhibiting overall motion of the satellite such as bending, torsion, and axial modes, and motions associated with appendages of significant mass, such as antennas, are considered to be significant dynamic response modes. For verification of test modes which are normalized to a unit generalized mass, test-to-test orthogonality of less than 0.05 for off-diagonal terms shall be a test goal. Validation of analytically computed modes shall be accomplished by cross orthogonality checks. Model-to-test orthogonality of 0.9 for terms on the diagonal, and 0.1 for off-diagonal terms, shall be a test goal. The modal survey shall monitor secondary modes of vibration represented by subsystems such as small tanks, batteries, and local support panels.

This shall be achieved by mounting accelerometers on selected secondary structures during modal search to identify frequency and modal energy content.

The modal survey shall use flight-quality satellite structure. Mass simulated components are acceptable, except for components/subsystems of significant mass with frequencies below 50 Hz. which require accurate dynamic simulation. The modal survey shall utilize a multi-shaker sine-dwell technique, or an approach of equivalent accuracy.

For vehicle systems where movements of mechanical devices and/or thermal/structural displacements of vehicle deployables or other subassemblies create dynamic movements of the vehicle (or components) which are critical to on-orbit performance, tests shall be performed which simulate such disturbances. The associated responses of critical components or subassemblies shall be measured to verify the effects on vehicle performance.

6.4.2 Acoustic Test.- An acoustic qualification/acceptance test shall be performed on the flight vehicle configured in its ascent configuration. The purpose of this test is to qualify the basic structure, all mechanisms, booms, antennae, solar arrays and other such subassemblies to the ascent vehicle qualification/acceptance level acoustic excitation and to provide a test bed for the acoustic qualification/acceptance of selected components. The vehicle shall be mounted on a simulated ascent vehicle adapter structure and instrumented to monitor the resulting random vibration response levels of the components and subassemblies. The random vibration response levels shall be used to verify the adequacy of the specified component test levels and to provide a basis for the development of individual random vibration test levels which need to be tailored for items not falling into the general environmental specification requirements. The lower cut-off frequency shall be established at an appropriate value for each specific vehicle, at not higher than 50 Hz or lower than 20 Hz.

6.4.3 Pyro Shock Survey.- Pyroshock survey tests shall be performed on a Structural Test Vehicle which satisfactorily simulates the flight vehicle, for vehicle/payload systems incorporating pyrotechnic devices. All pyrotechnically actuated release mechanisms shall be fired and the resulting shock levels shall be measured. A primary objective of this testing is to provide a pyroshock database to verify the adequacy of the component pyroshock test requirements in the general environmental specification. It will also provide a basis for verifying various shock interface environments such as those which exist at the booster interface.

6.4.4 Pyro Release Tests.- When applicable, pyro shock tests shall be performed on the vehicle immediately after the qualification/acceptance acoustic test. The test series shall include firing all pyrotechnic actuated release devices and verification of release and first motion. This test will subject the vehicle to the pyro shock levels associated with all spacecraft pyro events utilizing flight type explosive detonators and will verify that all mechanisms function properly after being exposed to the qual/acceptance acoustic levels. After completion of the pyro shock exposure the actuation mechanisms shall be functionally tested for damage detection or degradation of operation from the acoustic environment and pyrotechnic shock tests. This test will normally be accomplished in conjunction with the Pyro Shock Survey Tests.

6.4.5 Random Vibration Tests.- Random vibration environments and associated tests are principally formulated to provide vibration environment design and test requirements for components and subsystems. These design and test criteria enable establishment of a key basis for achieving component structural strength qualification and acceptance of such equipment. Such tests are especially pertinent where long lead time deliveries are necessary, and assurance is needed that the equipment can survive higher assembly system-level dynamic testing. Considerable care and foresight are needed in establishing maximum predicted vibration environments. However, it should be recognized that vibration environments do not necessarily include all of the maximum loading conditions that the item will endure.

(NASA/GSFC requirement))
6.4.6 Sine Vibration Tests.- Sine vibration environments and associated tests will be considered for applicability to simulate any potentially critical sustained periodic mission environment or to satisfy other requirements such as loads or shock.

6.5 VENTING TESTS

Where venting characteristics for components or assemblies cannot be analyzed with a reasonable level of confidence, venting tests simulating critical operational conditions shall be performed to verify the analyses specified in Section 5.5.

6.6 THERMAL CHARACTERISTICS TESTS

Flight Article thermal response characteristics shall be verified in conjunction with the system level thermal vacuum test performed to demonstrate component and assembly workmanship. The test shall be suitably instrumented to record data usable in updating, if necessary, the thermal math model utilized in performing the analyses specified in Section 5.6.

6.7 FRACTURE CONTROL INSPECTIONS

Inspection of fracture critical components shall be performed in accordance with the requirements generated by the fracture control analysis specified in Section 5.7. When feasible, such inspections shall be accomplished immediately after fabrication of the components or at the earliest subassembly level.

6.8 DEPLOYMENT & SEPARATION TESTS

Mechanical/deployable assemblies shall be tested under conditions simulating in-service operations, as agreed upon with the Procuring Agency. These tests shall verify the analyses specified in Section 5.8 and that adequate margins exist in regard to strength, power available, life, and clearances (whichever are critical) while undergoing the dynamics of their actual operations. Similarly, for systems involving separation maneuvers, separation test(s) shall be conducted to verify the same factors in regard to the separation system. Quasi-static or component testing may, in part, be adequate for verification of some of the deployment and separation subsystem characteristics.

7.0 DOCUMENTATION REQUIREMENTS

Reports shall be prepared which document the structural development plans, criteria/requirements, analyses and tests employed in the verification of the structural integrity of the vehicle/payload. The reports may be in the approved Engineering Memorandum (EM) format for the program, or in the standard LMSC report format as utilized on the program for formal data submittal. The sources of data utilized, the analytical and test methods employed, and assumptions used shall be defined. References cited in the reports which are not readily available shall be submitted to the University of Michigan with the reports; or, when practicable, included as Appendices to the reports.

The documentation shall include the following, as agreed upon with the University of Michigan: (1) Structural Development Plan; (2) Structural Design Criteria & Loads;* (3) Mass Properties; (4) Stress Analyses; (5) Structural Dynamics Analyses; (6) Thermal Analyses; (7) Structural Tests; (8) Dynamics Tests; (9) Fracture Control Analyses and Tests; (10) Thermal Tests; (11) Mechanisms Analyses and Tests; and (12) Strength Summary & Operating Restrictions. All documentation shall be in conformance with LMSC Division and Departmental policy directives and shall be issued with proper approval signature endorsements included on the title pages.

<u>* Note:</u> This document, LMSC/F440063, when issued with all "TBDs" completed and inserted in the document, will constitute completion of this requirement.

7.1 STRUCTURAL DEVELOPMENT PLAN

A plan for the structural engineering development and qualification shall be prepared which describes the total program for design/analysis and test for verification of structural adequacy, and includes schedules for its accomplishment. The plan shall definitize the analysis, test, and documentation requirements specified in Sections 5, 6 and 7 that are applicable to MUADEE and confirmed with the University of Michigan. The plan shall constitute a major input to the program's "Program Plan," or its "System Engineering Management Plan (SEMP) which may be prepared in compliance with such requirements as MIL-STD-499 (Appendix B, Reference 45), whichever is prescribed for MUADEE. Revisions shall be issued as necessary to reflect changes in objectives, requirements, design characteristics, and/or operational plans.

7.2 STRUCTURAL DESIGN CRITERIA AND LOADS

Refer to the "Note" under Paragraph 2.7.

7.3 LAUNCH VEHICLE CHARACTERISTICS

In the event that a "Payload Users' Manual" is not available, or is incomplete, the physical characteristics of the launch vehicle which are significant to the design of the payload vehicle structure shall be described and controlled by appropriate documentation, procedures, and policies. The physical description in the documentation shall include but not be limited to the following: vehicle dimensions; station locations; unit weights; weight distributions; centers of gravity; distribution of mass, inertia, and stiffness; vehicle modal data; and detailed configuration dimensions. The detail and accuracy of the documentation shall be sufficient to provide the basis for the structural analysis of the spacecraft and its interface with the launch vehicle.

7.4 INTERFACE CHARACTERISTICS

Physical and functional interfaces of the spacecraft structure with other customer furnished components, assemblies, systems, liquids, and gases shall be identified in interface control documentation. In conjunction with the University of Michigan, methods for controlling and accounting for interfaces shall be defined.

7.5 DESIGN ANALYSES

Reports shall be prepared on design/analyses performed to verify structural adequacy, per the requirements of Section 5. These reports shall be divided logically by subject and shall include loads analyses, stress & deformation analyses, structural dynamic response analyses, venting analyses, thermal analyses, fracture control analyses, mechanisms and separation analyses, and meteoroid and space debris protection analyses. Unless specifically directed by Program requirements, the SM 120 (Appendix B, Ref. 5) shall be used as a guide in coverage of strength analysis requirements, presentation format, and checking responsibility.

7.6 TEST PLANS, PROCEDURES & RESULTS

Test plans, procedures, and reports of results shall be prepared for all tests performed in accordance with Section 6. Test plans shall include a description of the test purpose, articles, requisite data, instrumentation, setup, conditions, accept-reject criteria, and a description of test results documentation. For flight tests, each test plan shall show how the test data will be extrapolated and interpreted in terms of design requirements when the test is conducted in a non-cooperative natural environment. Detailed test procedures shall be prepared by the test organization and approved by the cognizant design/analysis engineer.

The final test reports shall include the test results, conclusions, and recommendations; also, in case of failure, the report shall describe the failure, the failure condition, cause of failure, and the corrective action taken.

7.7 INSPECTION AND REPAIR

Reports shall be prepared on inspections accomplished in accordance with Section 6.7, and in the event of any repairs of the vehicle/payload. These reports shall include: descriptions of the techniques for inspection of the structure for the purpose of locating hidden defects, deterioration, and fatigue effects; and repair and replacement instructions, modified as necessary on the basis of flight-test experience.

7.8 STRENGTH SUMMARY & OPERATING RESTRICTIONS

A final Strength Summary and Operating Restrictions report shall be prepared which summarizes key strength margins of safety and any applicable vehicle operating restrictions. Significant updates to the stress analyses, which may have been necessitated after the main body of stress analyses was completed, may be included in this document in lieu of revising previously issued bulky documents for relatively small update changes. This document shall be the appropriate location for final identification of all structural integrity documentation, including discussion of their current status.

7.9 OPERATIONAL MONITORING MEASUREMENTS

Program plans normally include provisions for some level of monitoring of operational environments, mechanisms performance, and thermal control performance. This is accomplished by recorded measurements of accelerations, loads, pressures, temperatures, stresses, and/or critical deformations during preparation, handling and testing, and during ascent and orbital operations of the vehicle/payload. Operational monitoring reports shall define the objectives, parameters to be measured, describe the recording and monitoring systems, and the analyses planned, for evaluating structural, thermal control, and mechanism adequacy during vehicle operations.

APPENDIX A: STRUCTURAL CRITERIA GLOSSARY

Abort - A termination of a mission due to malfunction or failure.

Acceptance Tests - Tests performed on flight hardware to verify workmanship.

Ascent - (see Life Phases)

Assembly - A combination of two or more components that function as a discrete element of a system (see Component and System).

Atmospheric Flight - (see Life Phases)

Buffet - A repeated loading of a structure by an unsteady aerodynamic flow.

Burst Pressure - (see Pressure)

Component - A separate element, member or part of an assembly (see Assembly and System).

Condition - A phenomenon, event, time interval, or combination thereof to which the space vehicle (payload) is exposed. (see Design Condition)

Creep - A time-dependent deformation under load and/or thermal environment which results in cumulative, permanent deformation.

Critical - The extreme value of a load or stress, or the most severe environmental condition imposed on a structure during its service life. The design of the structure is based on an appropriate combination of such critical loads, pressures, stresses, and conditions.

Critical Flaw Size - The size of a flaw (a_{cr}) which, for given applied stresses and environment, causes unstable propagation of a flaw (see Flaws and Fracture)

Cyclic Flaw Growth Rate - The change in flaw size (a) per load cycle (n); da/nN.

Cryogenic Temperature - A temperature below (about) -100 degrees. C.

Design Condition - A condition which controls structural design and which may involve a specific point in time, or integrated effects over a period of time, in terms of physical units such as pressure, temperature, load, etc. (see Condition)

Design Factor - A multiplying factor applied to limit load or pressure for special purposes in addition to those normally included.

Design Temperatures - Temperatures of the structure when it is subjected to critical combinations of loads and pressures. For equipment item acceptance testing when not subjected to loads or pressures the worst case thermal model temperature prediction +20° F shall be utilized as design and test temperature (on the hot side, -20° F on the cold side). Similarly, for equipment item qualification testing the test temperature shall be +38° F above the predicted design temperature (hot side, -38°F on the cold side).

Deterministic - The process by which values are selected on the basis of known or assumed discrete data and not random. (see Probabilistic)

Detrimental Deformations - Displacements that cause contact or misalignment between adjacent components, which jeopardize the proper functioning of equipment, endangers personnel, or reduces the ability to ensure flight-worthiness below acceptable levels.

Divergence - A non-oscillatory instability which occurs when the external upsetting moments exceed the internal structural restoring moments within the system.

Elastic Mode - Same as Vibration Mode.

Emergency Condition - A loading, temperature, event, or combination thereof which exceeds specified limit conditions resulting from malfunction or other abnormal event.

Entry - (See Life Phases).

Environments - (1) NATURAL ENVIRONMENT: External conditions that exist in nature independent of the vehicle, such as temperature, pressure, radiation, winds, gusts, precipitation, meteoroids, earthquakes, and dust.

(2) MAN-MADE ENVIRONMENT: External conditions made by man that exist independent of the vehicle, such as sonic booms, explosions, and air contaminants.

(3) INDUCED ENVIRONMENT: Conditions created by the vehicle or its systems or by the response of the vehicle to the natural environments.

Factor of Safety (FOS) -

(1) DESIGN FOS: The multiplying factor applied to limit load (or pressure) to obtain ultimate, yield, or proof load (or pressure). (see Load Types)

(2) ALLOWABLE FOS: The ratio of vehicle/payload allowable yield or ultimate load to Limit Load. This factor is used as a measure of the actual load capability before yield or failure occurs.

Fail-Safe - A design philosophy under which failure propagation is so limited that the failure of any single structural component will not degrade the strength or stiffness of the remainder of the structure to the extent that the vehicle/payload cannot complete the mission at the specified limit load.

Failure - A rupture, collapse, excessive wear, or any other phenomenon resulting in an inability to sustain ultimate loads, pressure and environments (see Load Types).

Fatigue - In materials (structures), the cumulative irreversible damage incurred by cyclic application of loads (pressures) in given environments. Fatigue can initiate and extend cracks which degrade the strength of materials and structures. (see Flaws).

Flaws or Flaw-Like Defects - Defects which behave like cracks that may be initiated during material production, fabrication or testing, or are developed during the service life of a part. (see Fatigue).

Fracture Control - The rigorous application of load spectra analysis, stress analysis, quality assurance management, manufacturing, and operation technology dealing with the understanding and prevention of flaw propagation leading to catastrophic failure.

Fracture Critical Part - Any structural part, the fracture of which could threaten the safety of personnel.

Fracture Mechanics - An engineering discipline which describes the propagation behavior of flaws or flaw-like defects in materials under stress. (see Flaws and Fatigue).

Fracture Toughness - An inherent property of material which reflects the material's resistance to fracture. In fracture mechanics analysis, failure is assumed imminent when the applied stress intensity factor (K_i) is equal to or exceeds fracture toughness (K_{IC}).

Initial Flaw Size - The maximum flaw size (a_i) as defined by proof test or non-destructive evaluation, which is assumed to exist for the purpose of performing a fracture mechanics evaluation.

Initial Flaw - A flaw that exists in a part before it is subjected to applied loads or environmental effects.

Interface - The common boundary between components, assemblies, systems or support equipment of a vehicle /payload. An interface may be physical, functional, or procedural.

Life Phases - Subdivisions of vehicle flight and handling phases which are characterized by a related set of design conditions. Two categories of life phases may be identified: (1) those related to flight operations, including prelaunch, launch, ascent, and orbit; and (2) those related to ground operations, including manufacturing, storage, refurbishment, transportation and ground handling. Typical phase definitions are as follows:

(1) MANUFACTURING PHASE: The interval beginning with the manufacture of vehicle /payload hardware and terminating when the vehicle /payload and/or its systems, assemblies, or components are accepted for shipment from the manufacturing facility to the launch site or storage area. Manufacturing includes receiving, inspection, fabrication, assembly, and checkout operations.

(2) STORAGE PHASE: The interval during which a vehicle /payload and/or its systems, assemblies, or components are stored in an inactive condition.

(3) REFURBISHMENT PHASE: An interval during which a vehicle /payload and/or its systems, assemblies, or components are repaired, replenished, inspected, or tested.

(4) TRANSPORTATION AND GROUND HANDLING: Intervals and events during which the vehicle /payload and/or its systems, assemblies, or components are handled, transported, or erected. Each transport interval begins when the vehicle / payload is accepted or certified for shipment and terminates when shipment is received at its destination. Ground handling includes towing, hoisting, supporting, reorienting, carrying, erecting, jacking, and mooring.

(5) PRELAUNCH PHASE: The interval beginning with completion of vehicle installation on the launch pad and terminating with engine ignition.

(6) LAUNCH PHASE: The interval beginning with engine ignition and terminating when launch transients have decayed to negligible values.

(7) ASCENT PHASE: The interval beginning after launch transient decay and terminating at insertion into orbit.

(8) ORBIT PHASE: The interval beginning with orbit insertion. It could also include rendezvous, docking, undocking, cargo transfer, and/or mechanical operations in space.

(9) ENTRY PHASE: The interval beginning with the completion of deorbit retro impulse and terminating after the transition of the entry vehicle to aerodynamically controlled flight.

(10) LANDING PHASE: The interval beginning with touchdown of the vehicle or the Orbiter, and terminating after the landing or taxi run. Landing includes touchdown, landing roll, braking, and taxiing, as may be applicable.

Load Factor - The ratio of the inertial forces acting on the mass of the body to the weight of the body.

Load Redistribution - The changes in load distribution due to elastic or inelastic deformation of the vehicle /payload and/or its system, assemblies, or components.

Load Spectrum - A representation of the cumulative static and dynamic loadings anticipated for a structural component or assembly under all expected operating environments.

Load Types - External Forces applied to the vehicle /payload and/or its systems, assemblies, or components. Load types are as follows:

(1) LIMIT LOAD (OR PRESSURE): The maximum load (or pressure) expected to act on a structure over its service life. Limit loads include static, quasi-static, dynamic, and impulse loads.

(2) DESIGN YIELD LOAD (OR PRESSURE): The product of the limit load (or pressure) and the design yield factor of safety.

(3) ALLOWABLE YIELD LOAD (OR PRESSURE): The load (or pressure) below which no deformation of the structure that will jeopardize the mission will occur in the specified operating environment.

(4) DESIGN ULTIMATE LOAD (OR PRESSURE): The product of the limit load (or pressure) and the design ultimate factor of safety.

(5) ALLOWABLE ULTIMATE LOAD (OR PRESSURE): The load (or pressure) below which no rupture, collapse, or other mode of failure of the structure will occur in the specified operating environment.

(6) PROOF LOAD (OR PRESSURE): The load (or pressure) above limit load (or pressure) applied to a structural component or assembly as the basis for evaluating quality of materials and workmanship.

(7) DYNAMIC LOAD: A transient load from sources internal or external to the structure associated with vibrations, shocks, oscillatory motions, and acoustic effects.

(8) STEADY-STATE LOAD: A load of constant magnitude and direction with respect to the structure.

(9) QUASI-STATIC LOAD: A statically applied load which is considered to realistically represent the maximum value of a time-varying load.

(10) IMPULSE LOAD: A suddenly applied pulse or step change in loading in which the duration is small compared to the period of the highest structural mode which contributes significantly to the total load.

Margin of Safety (MS) - The increment by which the allowable load (or stress) exceeds the design load (or stress) for a specific design condition (i.e. yield or ultimate), expressed as a fraction of the design load (or stress).

$$MS = \frac{\text{Allowable Load (or Stress)}}{\text{Design Load (or Stress)}} - 1$$

The margins so determined are used as final indicators of available strength after all other design characteristics, conditions, and factors have been accounted for at each service condition.

Modal Test - Tests performed to determine the mode shapes, frequencies and damping coefficients of the satellite and to provide experimental data for validation of the dynamic models.

Model Uncertainty Factor (MUF) - A factor to account for the unknown vehicle dynamic response to the applied load environment. This factor is applied to the raw computed applied loads data to define limit loads.

Pressure - A force applied to a structure prescribed over a unit area. Pressures can be external, internal, constant or varying. Pressure conditions are as follows:

(1) BURST PRESSURE: The maximum differential pressure at which an internally pressurized component ruptures.

(2) COLLAPSE PRESSURE: The maximum differential external pressure that a component can sustain without compression instability failure.

(3) HEAD PRESSURE: Static head pressure is the pressure at any point below the liquid level in a pressure vessel due to height of the column of liquid in a gravity field. Dynamic head pressure is the additional pressure caused by acceleration.

(4) LIMIT PRESSURE: The maximum differential pressure that can be expected to occur in service under the expected operating environments. Limit pressures include maximum expected operating pressure, transient pressure, and head pressure.

(5) MAXIMUM EXPECTED OPERATING PRESSURE (MEOP): The maximum pressure at which the system or component is actually expected to operate. MEOP includes the effects of worst case tolerance on environmental controls, relief valves, pressure regulators, etc.

(6) MAXIMUM OPERATING PRESSURE (MOP): The maximum pressure at which a system or component could operate including the effects of credible failure conditions.

(7) MINIMUM OPERATING PRESSURE: The minimum pressure applied to a pressure vessel by the pressurizing system with the pressure regulators and relief valves at their lower limit and with the minimum flow rate.

(8) PROOF PRESSURE: The pressure that components must sustain to give evidence of satisfactory workmanship and material quality, and is used to establish the maximum undetected flaw size (in fracture control). It is equal to the product of the limit pressure and the proof factor. (see Proof Load).

(9) TRANSIENT PRESSURE: Time-dependent pressure in which the characteristic time of fluctuation is comparable to significant dynamic time constants of the structure

and vehicle systems; e.g., valve opening and closing, pump surge, check or relief valve flutter, engine thrust transients, and fluid slosh.

(10) WORKING PRESSURE (Nominal Operating): The nominal operating pressure applied to a pressure vessel by the pressurizing system with the pressure regulators and relief valves at their nominal settings and with a nominal fluid flow rate.

(11) YIELD PRESSURE: The differential pressure below which no detrimental deformation will occur in the specified operating environments. It is equal to the product of the limit pressure and the yield factor of safety. (see Design and Allowable Yield Load).

(12) ULTIMATE PRESSURE: The maximum differential pressure below which no structure (vessel) will rupture or collapse in the expected operating environments. It is equal to the product of the limit pressure and the ultimate factor of safety. (see Design and Allowable Ultimate Load).

Pressure Vessel - A container designed primarily for pressurized storage of gases or liquids and:

(1) containing stored energy of 14,240 foot-pounds (0.01 lbs. of TNT equivalent) or greater based on adiabatic expansion of a perfect gas,

OR

(2) will experience a design limit pressure greater than 1,000 psia,

OR

(3) contains a fluid in excess of 15 psia. which will create a hazard if released.

Pressurized Structure - A structure designed primarily to carry vehicle/payload loads, but which may also be subjected to internal pressure.

Probabilistic - The process by which values are derived on the basis of statistical inference, as opposed to deterministic.

Proof Factor - A multiplying factor applied to either limit load or limit pressure to obtain either proof load or proof pressure.

Proof Test - The test of a flight structure at proof load /pressure which will give evidence of satisfactory workmanship and material quality or will establish the maximum undetected flaw size. (see Proof Load / Pressure)

Protoqualification Test - The test of the first production unit to a more severe level than acceptance test, but less severe in either level or duration than the conventional qualification test. The testing consists of the same types of testing and test sequences as is used in the qualification testing. These tests are conducted on flight vehicle/components.

Qualification Tests - Tests conducted on flight-quality components or assemblies at load levels sufficient to demonstrate that design requirements have been achieved.

Random Vibration - The non-periodic motion of a structure caused by acoustical and/or mechanical forcing functions.

Safe-Life Design - A design philosophy under which propagation of undetected flaws to failure will not occur in the expected operating environments during the specified service life of the structure or between inspection intervals. (see Fatigue and Fracture Control).

Satisfactory Containment - A part is satisfactorily contained if the separation of the part from its support, or the fracture separation of any portion of the part, cannot lead to a release of fragments that are a hazard to personnel

Satisfactory Redundancy - A design has satisfactory redundancy if failure of a single element when subjected to limit loads will not lead to catastrophic failure of the remaining structure, or to the occurrence of any other hazardous event. (see Fail-Safe).

Service Life - The interval beginning with manufacture of a vehicle /payload and ending with completion of its specified missions.

Special Nondestructive Evaluation - The formal inspection of parts using nondestructive procedures involving the use of techniques and/or equipment that exceed common industrial inspection standards.

Standard Nondestructive Evaluation - The formal inspection of parts using nondestructive procedures consistent with common industrial standards. These standard procedures include visual, 10X magnified visual, dye penetrant, eddy-current, magnetic particle, ultrasonic, and x-ray.

Stiffness (Rigidity) - Resistance to elastic deformation under an applied force.

Stresses - An internal force per unit area. The following types occur in structures:

(1) APPLIED STRESS: Stresses resulting from applied loads, pressures and environments. These can be in the form of axial, bending, shear, and /or torsion stresses.

(2) RESIDUAL STRESS: Stress that remains in the structure due to processing, fabrication, or prior loading.

(3) THERMAL STRESS: Stresses resulting from temperature gradients and differential thermal deformations within or between structural components, assemblies, or systems.

Stress Corrosion Cracking - The initiation and/or propagation of cracks due to combined action of applied stresses and environmental effects.

Stress Intensity Factor $K_{(i)}$ - A calculated quantity which is used in fracture mechanics analyses as a measure of the stress-field intensity near the tip of an idealized crack. (see Fracture Toughness).

Structure - All components and assemblies designed to sustain loads or pressures, provide stiffness and stability, or provide support or containment.

(1) PRIMARY STRUCTURE: Structure required to transmit acceleration and/or externally applied loads throughout the assembly.

(2) SECONDARY STRUCTURE: Structure not required to transmit acceleration and/or externally applied loads throughout the assembly. Component packages and component supporting structure are elements considered to be secondary.

Structural System - A major combination of components and assemblies that functions as a unit. (see Assembly and Component).

Structural Test Vehicle (STV) - A dedicated flight -quality assembly designated for structural testing. This mode of structural verification testing must be carefully evaluated to assess the confidence that flight vehicle response to the design environments can in fact be simulated with the STV as it is designed and fabricated. An STV with mass simulated components does not provide satisfactory flight vehicle simulation for acoustic and pyroshock survey tests.

Threshold Stress Intensity Factor **(K_{TH})** - The maximum value of stress intensity factor for a given material at which no environmentally induced crack growth will occur in the specified loading environment.

Ultimate Strength - Corresponds to the maximum load or stress that an unflawed structure or material can withstand without incurring rupture or collapse.

Yield Strength - Corresponds to the maximum load or stress that an unflawed structure or material can withstand without incurring detrimental deformation.

APPENDIX B: BIBLIOGRAPHY & REFERENCES

The following documents, or sections of documents, contain information and/or data applicable to the present document and augments this document in matters pertaining to all elements of the vehicle/payload structural design criteria. In the event of conflicting criteria statements between this document and the following References refer to the Section 1.4 requirements.

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APPENDIX C: LAUNCH VEHICLE ENVIRONMENT

The Delta II 7925 Launch Vehicle is the booster system currently under study for use on the MUADEE mission. The best published data for launch, ascent, and separation environments, and booster interface data, is contained in the Delta vehicle Payload Planners Guide (Appendix B, Reference 53). McDonnell Douglas advises that an update is in progress, as of the date of this document, which is expected to be released in the next few months. Currently we are using the Ref. 53 document, with some changes as provided in telecons with the McDonnell Douglas Delta Program Integration Office engineers.

When the new issue of Ref. 53 becomes available, and the conceptual studies have progressed further, this Appendix C will be modified to include all pertinent data from the Planners Guide so as to have the related criteria applicable to spacecraft design contained under one cover.

3/15/94

FLIGHT SOFTWARE REQUIREMENTS SPECIFICATION

FOR

THE MUADEE SPACECRAFT PROGRAM

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TABLE OF CONTENTS

1.0 INTRODUCTION

1.1 Identification of Document

This is a Software Requirements Specification (SRS) of the Mars Upper Atmosphere Dynamics, Energetics, and Evolution (MUADEE) Mission Flight Software (FS) which includes four Computer Software Configuration Items (CSCIs):

Start Up Software CSCI	STARTUP_SW_CSCI (SUS)
Command & Telemtry Software CSCI	COMMAND_TLM_SW_CSCI (MSS)
Mission Support Software CSCI	MISSION_SUPP_SW_CSCI (MSS)
Backup Safemode Software CSCI	BACKUP_SAFEMODE_SW_CSCI (BSS)

This document contains the performance, design and interface requirements for the MUADEE FS, including a general description of its major functions, the functional requirements, a description of the MUADEE FS internal data interfaces, and a general specification of MUADEE FS/MUADEE hardware data interfaces.

The MUADEE FS resides in the MUADEE flight computer and supports MUADEE mission performance as specified in this section. As its primary function, the MUADEE FS supports the Attitude and Control Subsystem to provide control of vehicle attitude and provide a stable platform for the science instruments. The MUADEE FS interfaces with attitude control hardware flight components of the spacecraft to: receive data, process data as required, and issue control commands. The FS also performs additional tasks to support the MUADEE mission: supporting telemetry and command functions; monitoring intermediate software values for safemode protection and triggering safemode macros should those values exceed nominal ranges; monitoring hardware data for safemode protection and triggering safemode macros should those values exceed nominal ranges; gathering data from the payload computer and passing it to telemetry. The MUADEE FS interfaces with the ground segment, the Michigan Test and Operations Control Center (MTOC).

1.2 Scope of Documentation

This SRS is applicable to the SUS, CTS, MSS, and BUS CSCIs of the MUADEE Flight Software System.

1.3 Purpose and Objectives of Document

The purpose of this document is to provide traceable requirements from which the software for the CSCIs is designed. The objective of this document is to provide documentation showing requirements necessary to develop and test the CSCIs to produce a product that meets all the requirements.

1.4 Document Status and Schedule

This SRS is in a preliminary state and is distributed for the MUADEE pre-proposal data package.

2.0 RELATED DOCUMENTATION

TBD

3.0 EXTERNAL INTERFACE REQUIREMENTS

The External Interfaces for the MUADEE Flight Software are limited to the Command and Data Handling (C&DH) System interface.

4.0 REQUIREMENTS SPECIFICATION

The MUADEE FS is organized into four Computer Software Configuration Items (CSCIs): The Start-Up Software (SUS), the Command and Telemetry Software (CTS) the Mission Support Software (MSS), and the Backup Safemode Software (BSS). For this document, the CTS will be discussed in the MSS section since they are both operating during the same operating mode. A separate bootstrap CSCI and real time operating system CSCI shall be provided by the computer vendor. The SUS shall always begin when the flight computer begins operation, after hand-off from the bootstrap CSCI. The MUADEE MSS shall begin processing only when commanded. The BSS shall begin processing based on ground command or as a result of safemode entry, when the primary random access memory (RAM) based safemode system fails. The real time operating system works with all three MUADEE FS CSCIs.

4.1 Process and Data Requirements

The process and data requirements for each CSCI are specified in this section. Section 4.1.1 specifies the Start-Up Software (SUS) CSCI, section 4.1.2 specifies the Mission Support Software (MSS) CSCI and Command and Telemetry Software (CTS) CSCI, and section 4.1.3 specifies the Backup Safemode Software (BSS) CSCI. For section 4.1.2, all references will be made to the MSS which also includes the CTS.

4.1.1. Start-Up Software

The Start-Up Software (SUS) is a CSCI which activates whenever the computer is activated by hardware command, after the bootstrap CSCI has completed. The SUS is in the programmable read only memory (PROM) section of the computer memory. The detailed requirements are found in the following subsections.

4.1.1.1. Start-Up Software Initialization

Upon initial software start-up the SUS shall initialize its variable data base, command the C&DH to output telemetry and synchronize the output of the software values to the C&DH Telemetry Control module.

4.1.1.2. Interrupt Handler

The SUS shall be driven by the 100 Hz interrupt supplied by the C&DH.

4.1.1.3. Keep Alives

The SUS Keep Alive (KA) function shall calculate Keep Alive values to be sent to the monitoring hardware in the C&DH every 250 ms ± 100 ms. The KA value shall be generated by combining two non-zero numbers and outputting the result without storing the result or the intermediate values in memory.

4.1.1.4. Input

The SUS shall receive uplinks from the MTOC via the C&DH as 16 bit words. This uplink shall conform to CCSDS standards.

4.1.1.4.1. Acceptance Criteria

The SUS shall accept no RTC or memory block load until first receiving the correct security code word; following which it shall accept one, and only one, RTC or memory block load. Each RTC or memory block load shall be temporarily held in a buffer until the final word - the checksum word - has been received and the acceptance tests passed. If the acceptance tests are not passed then status words available to telemetry shall be updated to reflect the test failure and the temporary holding buffers shall be set to zero. A counter shall be kept of uplinks that were not preceded by the correct security code. This counter shall be available to telemetry.

4.1.1.4.2 Block Loads

The SUS shall process block load commands from the ground. It shall perform this function in the same manner as the CTS component of the MUADEE FS. The block load command functions shall accept blocks of 1 to 64 words into a temporary holding buffer. It shall perform acceptance tests on the block and provide an accept/reject flag for telemetry. This flag shall be latched on the first occurrence of a checksum failure, an attempt to write to a write-protected area of memory, or a failure to complete a load of the number of words specified in the header. The flag shall remain latched, following a failure, until cleared by ground command. If the block passes the tests, the block load function shall store the block in the area specified in the block load command header. This uplink shall conform to CCSDS standards.

4.1.1.4.3 Commands

The SUS shall process S/W Real Time Commands (RTCs). This uplink shall conform to CCSDS standards.

4.1.1.4.3.1 Memory Protect Command

The SUS shall process memory protect commands from the ground. These commands shall allow for the control of the computer Write Protect and I/O Write Protect Status of each block of memory.

4.1.1.4.3.1.1 Computer Write Protect Command

The computer Write Protect command shall configure the computer Write Protect function. The computer can still read from a protected block of memory.

4.1.1.4.3.1.2 I/O Write Protect Command

The I/O Write Protect command shall configure the computer I/O Write Protect function. An I/O Port can still read from a protected block of memory.

4.1.1.4.3.2 Memory Dump Command

The SUS shall process the memory dump commands from the ground. The memory dump command shall cause the contents of memory to be dumped to the telemetry downlink port. The command shall indicate a starting and ending 64-word block of memory. The memory will be dumped starting from the first word in the indicated starting block. The memory dump shall continue dumping from the starting block to the ending block and thence back to the starting block until either the SUS telemetry mode command is received or another memory dump command with a different dump range is received.

4.1.1.4.3.3 SUS Telemetry Command

The SUS telemetry command shall cause the SUS to switch from the memory dump format to the SUS format. If the SUS is already in SUS telemetry format then there shall be no response to the command.

4.1.1.4.3.4 MSS Initialization Command

The SUS shall process the MSS initialization command. This command shall cause the SUS to start the MSS initialization module and to suspend itself. The SUS shall have the option of transitioning directly to the MSS upon start-up without requiring a ground command. In either ground command or automatic mode, the MSS initialization function shall have the ability to execute a specified macro sequence to configure the MSS to a specific operating mode.

4.1.1.5. Output

The SUS shall, upon power-up, send telemetry values to the C&DH that provide basic SUS hardware and flight software health and status. The SUS shall support two formats: the aforementioned health and safety format and a memory dump format, each of which shall be identified by a unique format identifier.

4.1.1.5.1. SUS Telemetry

The SUS S/W Telemetry format shall be output to the S/W dump port. The first word of the SUS S/W Telemetry format shall be a format identifier to distinguish it from the memory dump.

4.1.1.5.2. Memory Dump

The SUS Memory Dump format shall be output to the S/W dump port. The first word of the SUS Memory Dump format shall be a format identifier to distinguish it from the SUS S/W Telemetry format.

4.1.1.5.3. Keep Alives

The SUS Keep Alives shall be issued through port 1 to the C&DH Watchdog timer.

4.1.1.6. SUS Safemode Support

The safemode processing shall test sensed or calculated values against limits set in the FS constant data base. A safemode response shall be activated when a tested value exceeds the data base limit for another data base limit consecutive tests. All data base values shall be user configurable.

4.1.1.6.1. Memory Parity Error Test

The Memory Parity Error Test function shall continually test for memory parity errors in all available memory, while the SUS software is running. It shall be able to test memory within 30 seconds. For each parity error detected, it shall store the memory address where the last error detected occurred, overwriting the previous data. During software start-up the address data shall be zeroed. For each memory parity error test pass, it shall count the number of parity errors and store the total count. The contents of this counter and the last parity error address shall be made available to the SUS telemetry.

4.1.1.6.2. Write Protect Violation Interrupt

The Write Protection Violation detection shall be implemented as a response to the Write Protection Violation Interrupt. The interrupt response module shall increment the Write Protect Violation Counter whenever a Write Protect Violation Event is detected. When the counter exceeds a data base limit the Write Protection Violation Flag shall be set and the SUS Keep Alive processing shall be halted.

4.1.1.7. Restart Capability

If the computer is halted and restarted, the SUS shall self-initialize and begin operation in a known and repeatable configuration. This capability will exist no matter what state the SUS was in when the computer was halted.

4.1.2. Mission Support Software/Command & Telemetry Software

The Mission Support Software (MSS) is a CSCI which activates upon ground command or autonomously from the SUS CSCI. It interacts with the Command and Telemetry Software (CTS) CSCI to support mission operations. The detailed requirements are found in the following subsections and combines both CSCIs.

4.1.2.1. Mission Support Software Initialization

The MSS initialization processing shall be called one time by the SUS upon transition of control from the SUS to the MSS.

4.1.2.1.1. Reset all MSS Control Flags

The MSS Initialization processing shall assume computer control from the SUS. Upon assumption of control, it resets all of the MSS control flags.

4.1.2.1.2. Synchronize 10 Hz and 100 Hz Interrupts

MSS Initialization shall synchronize the 100 Hz processing with the 10 Hz processing, such that each 100 Hz cycle shall be synchronous with every 10th 10 Hz cycle.

4.1.2.1.3. Configure Initial MSS Processing

MSS Initialization processing shall allow the MSS scheduler to operate for 1 second such that each process can self-initialize. It then will optionally activate specific MSS control flags based on a macro sequence preloaded by the ground segment.

4.1.2.1.4. Transition of Control to MSS Schedule

After performing all initialization functions, MSS Initialization processing will turn over computer control to the Scheduler.

4.1.2.1.5. MSS Initialization Performance and Timing

MSS Initialization shall operate on a one time basis. It shall complete all functions within two seconds.

4.1.2.2. Executive and Support Processing

4.1.2.2.1. Interrupt Processing

4.1.2.2.1.1. 100 Hertz Interrupt

The 100 Hz processing shall be activated by the 100 Hz interrupt and shall be synchronized with the 10 Hz interrupt in such a fashion that the 100 Hz interrupt that starts the I/O processing is coincident with the first 10 Hz interrupt.

4.1.2.2.1.2. 10 Hz Interrupt

The MSS scheduler shall be activated by the 10 Hz interrupt.

4.1.2.2.2. MSS Scheduler

The Mission Support Software (MSS) CSC scheduler shall call each CSC in a specified order. The following discussion delineates the calling sequence and timing requirements that the scheduler shall observe.

4.1.2.2.2.1. Activated by Hardware Interrupt

The scheduler shall be activated with a 10 Hz hardware interrupt and shall be interruptable by the 100 Hz processing, which services the input/output functions. The 10 Hz processing shall always interrupt any on-going 1 Hz processing.

4.1.2.2.2.2. Module Called by Scheduler

Each module shall always be called by the scheduler. The module shall execute as specified by its corresponding control flag.

4.1.2.2.2.3. Suspended Lower Rate Processing

Processing segments interrupted by higher priority processing segments shall be temporarily suspended and then restarted when all higher priority segments are finished.

4.1.2.2.2.4. Timed Out Error Check

Before the start of a segment, a check shall be made on whether the preceding segment has completed processing in its allocated time. If not, a timed-out error indicator shall be incremented before the segment is started. The timed-out indicator shall be available for telemetry output. It shall also be made available to Safemode Processing. The indicator shall be reset only by MTOC command.

4.1.2.2.2.5. Performance and Timing

4.1.2.2.2.5.1. Processing Rate

The scheduler shall be invoked each time a 10 Hz interrupt is generated by the oscillator. The 100 Hz interrupt shall have precedence, and shall be used to schedule all I/O. A 1 Hz software counter (.i.e. ten 10 Hz cycles) shall be used to initiate the 1 Hz CSCs.

4.1.2.2.2.5.2. Gyro Data Relevancy

The gyro data shall be read and processed immediately after receipt of the 10 Hz interrupt. The read from the hardware shall be completed within 0.2 milliseconds.

4.1.2.2.2.5.3. Thruster Command Lag Interval

The thruster commands shall be issued in normal mission mode after the gyro data processing and vehicle control law CSCs have executed. These commands shall be issued no later than 30 milliseconds after receipt of the 10 Hz interrupt.

4.1.2.2.2.5.4. Attitude Determination Lag Interval

The attitude error as determined from the attitude sensors shall be computed during the 1 hz processing loop. Upon start of the 1 hz loop, the attitude error shall be made available to the vehicle control law within 0.2 seconds of the start of the new 1 hz interval.

4.1.2.2.3. Input

The data transfer process from sensors to the MUADEE FS shall be initiated by the 100 Hz processing but shall then operate independently of it in a Direct Memory Address (DMA) mode.

4.1.2.2.4. Output

The command transfer process from the MUADEE FS to the actuators shall be initiated by the 100 Hz processing but shall then operate independently of it in a DMA mode.

The telemetry transfer process from the MUADEE FS to the actuators shall likewise be initiated by the 100 Hz processing but shall then operate independently of it in a DMA mode. The order of data presented to the telemetry dump channel shall be determined by the telemetry format selected.

4.1.2.2.5. Keep Alives

The MSS Keep Alive (KA) function shall calculate Keep Alive values to be sent to the monitoring hardware in the C&DH every 250 ms ± 100 ms. The KA value shall be generated by combining two non-zero numbers and outputting the results without storing the results in memory.

4.1.2.2.6. Vehicle Time Word Update

The software shall continue to update the vehicle time word based on the last valid reading from the ground update.

4.1.2.2.6.1. Vehicle Time Word Reset

The vehicle time word shall be capable of being reset by SPC to a resolution of 0.1 seconds.

4.1.2.2.7. Command Handler

The command handler shall execute Stored Program Commands (SPCs) from computer memory as a function of vehicle time and location in memory; it shall accept uplinked real time commands (RTCs); and shall load uplinked data into computer memory. Detailed input, processing and output requirements are defined in subsequent paragraphs.

4.1.2.2.7.1. Inputs

The MSS shall receive uplinks from the MTOC via the C&DH as 16 bit words.

4.1.2.2.7.1.1. Acceptance Criteria

The Command Handler shall accept no RTC or memory block load until first receiving the correct security code word; following which it shall accept one, and only one, RTC or memory block load. Each RTC or memory block load shall be temporarily held in a buffer until the final word - the checksum word - has been received and the acceptance tests passed. If the acceptance tests are not passed then status words available to telemetry shall be updated to reflect the test failure and the temporary holding buffers shall be set to zero. A counter shall be kept of uplinks that were not preceded by the correct security code. This counter shall be available to telemetry.

4.1.2.2.7.1.2. Memory Block Loads

The Command Handler shall transfer memory block loads that have passed the acceptance criteria to the location in memory specified in the memory block load header.

4.1.2.2.7.1.3. S/W Real-Time Commands

The Command Handler shall place the pointer to the beginning of the temporary buffer containing the RTC in the mailbox shared by the Command Handler with the module that designed to process that RTC. Real time S/W Commands shall have priority over the Stored Program Commands.

4.1.2.2.7.1.4. H/W Real-Time Commands

The Command Handler shall place the accepted hardware RTC in the RTC hardware command buffer.

4.1.2.2.7.2. Outputs

The outputs from the Command Handler shall be comprised of status words to telemetry, hardware commands, and mailbox values to the various MUADEE FS modules.

4.1.2.2.7.2.1. Status Flags

The Command Handler shall maintain for telemetry status words that indicate all failure conditions, the current SPC being processed by the Timed and Conditional SPC processors, the number of hardware commands in the RTC and SPC hardware command buffers, and the state of all mailboxes shared with other MUADEE FS modules.

4.1.2.2.7.2.2. Mailboxes

The Command Handler shall maintain for telemetry the state of all mailboxes shared with other MUADEE FS modules. Each MUADEE FS module shall have two mailboxes associated with it - a command mailbox and a pointer mailbox. The pointer mailbox shall contain the pointer to the command to be executed by the module. The command mailbox shall contain the relative command number. The relative command number shall be calculated by subtracting the module command bias value from the command number.

4.1.2.2.7.3. Command Structure

The SPCs shall conform to CCSDS standards. Furthermore, it shall only accept SPCs with a recognizable command header. The Command Handler shall respond to unrecognized SPCs by ceasing to process SPCs and by placing the vehicle in the safemode that represents the next lowest operation mode. The structure of Real-time Commands shall differ from that of SPCs only in that they have no time tags.

4.1.2.2.7.4. Stored Program Commands (SPCs)

The Command Handler shall maintain multiple (10 or more) processors capable of executing sequences of SPCs. This processing function shall execute command lists as a function of vehicle time or an onboard instruction resulting from computation or decisions accomplished by software. Furthermore, software control constructs directed to the command handler itself shall be executed upon encounter without regard to vehicle time. The conditional SPC (CSPC) processor shall provide the capability to implement a minimum of nine special control constructs as follows:

1. Conditional SPC branch - Moves CSPC pointer to address designated within command packet.

2. Timed SPC branch - Moves Timed SPC (TSPC) pointer to address designated within command packet.

3. Timed wait - Renders the command handler processor, in which this command is executed, inactive until the onboard time word equals the designated value.

4. Expire - Activates a countdown timer to provide an elapsed time so that if the time expires, an override or disruptive SPC pointer move occurs.

4. Resume - Causes CSPC processor to delay command processing until an internal status flag indicates a process is complete.

6. Result - Causes CSPC processor to delay until status flag indicates success or fail and directs success or fail address in command packet to be processed.

7. Circle - Causes CSPC processor to cycle through a set of SPC pointers contained in command packet.

7a. Case - Causes CSPC processor to return to an SPC pointer previously selected via acquisition logic.

8. Repeat - Causes CSPC pointer to move to the same address where the repeat count is incremented by one, and when the count exceeds a specified count, the repeat complete address occurs.

9. Initialize repeat - Causes the repeat counter or the circle index, whichever construct is next encountered, to be reset to zero.

4.1.2.2.7.4.1. Execution Order

The order of execution of SPCs shall be determined solely by their location in memory.

4.1.2.2.7.4.2. Time Tags

The execution time of an SPC shall be determined by the time tag. When the vehicle time equals the time tag then that SPC shall be executed. If the time tag is absolute time then the execution time is set to be the time tag. If the time tag is delta time then the execution time is calculated to be the execution time of the previous command in that processor plus the delta time value. If the time tag is delta time and zero then the command is executed immediately. Stored commands having the same time tag shall be issued every 10 Hz cycle during the execution time specified. They shall be issued in the order they were loaded, up to a maximum of 10 commands per second.

4.1.2.2.7.4.3. SPC Rejection Conditions

An SPC shall be rejected, with appropriate status information, if the mailbox for the intended module is positive and non-zero. An SPC shall be accepted, with appropriate status information, if the mailbox for the intended module is zero. An SPC shall be accepted, with appropriate status information, if the mailbox for the intended module is negative (high-bit set) and the bit corresponding to the relative command number is zero. An SPC shall be rejected, with appropriate status information, if the mailbox for the intended module is negative (high-bit set) and the bit corresponding to the relative command number is one.

4.1.2.2.7.4.4. Conditional Commands

The Command Handler shall support commands that perform one of several actions dependent upon the value of a Module Status Flag (MSF), which is unique for each processing module to be activated. This flag shall be set by the Command Handler upon execution of one of a subset of the full S/W command repertoire to the 'Busy' state. The processing module executing the command shall be responsible for setting the MSF to either 'Success' or 'Fail' depending on the result of its command execution. The Command Handler shall be responsible for supporting conditional commands whose actions will be dependent upon the value of the MSF. Upon execution of one of these conditional commands, the Command Handler shall reset the MSF to the 'Not Busy' state.

4.1.2.2.8. Telemetry Control

Telemetry Control shall respond to commands sent to the Telemetry Mailbox by establishing any programmable format available in memory or by sending memory dump.

4.1.2.2.9. Science Instrument Communications

Science Instrument Communications shall schedule the transmission and receipt of Processor Interface Tables (PITs) between the three individual Data Processing Units (DPU) for each Instrument Package and the FS. The structure of these tables is TBD.

4.1.2.2.10 High Gain Antenna Control

High Gain Antenna (HGA) Control shall determine HGA pointing using on-board ephemeris for the spacecraft and Earth.

4.1.2.2.11 Thermal Control Law

The Thermal Control Law shall, when the Thermal Control Flag is set to 'on', monitor the thermistors on the vehicle and payload and send commands to maintain a nominal temperature on the payload.

4.1.2.2.12 Mass Storage Device Control

The Engineering/Science Mass Storage (ESMS) management function shall provide for management of the MUADEE on board solid state recorders.

4.1.2.2.12.1 ESMS Command Capability

The ESMS management function shall provide MTOC with the capability to initiate recording of science data on the mass storage device or sector designated as the Science Recorder and to initiate playback of data from any recorder.

4.1.2.2.12.2 ESMS Record Option Requested by SI

The record option will configure the FS to send the MTOC-specified command to the Science Recorder upon receipt of a record request through the Science Instrument interface.

4.1.2.2.12.3 ESMS Playback Option

The playback option will allow MTOC to send a series of commands to the designated recorder to control the playback, synchronized with ground receipt configuration and playback time window.

4.1.2.2.12.4 ESMS Single Command Initiates a Command Sequence

A single request will cause issuance of all commands in sequence with a MTOC-specified delta time between commands.

4.1.2.2.12.5 ESMS Management Processing rate

The ESMS management function shall operate at 1 Hz.

4.1.2.2.12.6 ESMS Compatibility with SPC Processors

The ESMS management function shall be compatible with the SPC processors. It shall be possible to manage the ESMS with an SPC processor independent of other SPC processor activity.

4.1.2.3. Attitude Reference

The Attitude Reference Processing shall provide the vehicle attitude, deriving it from information received from the Horizon Sensors, Star Scanner, and Sun Sensors; the vehicle position, deriving it from information received from the Vehicle Ephemeris and Accelerometers; the sun position, deriving it from the Sun Ephemeris and the Sun Sensors; the vehicle angular rate, deriving it from information received from the Control Gyros.

The attitude reference processing shall provide attitude and control for all modes of operation, Cruise, Capture, and Science Operations (Spin and Despun) modes.

Each module shall output its own version of the attitude, position, or rate. Thus any reference to zeroing the output of a module applies only to its version of that output.

4.1.2.3.1. Vehicle Angular Rate Reference

4.1.2.3.1.1. Control Gyro Processing

The Control Gyro Processing shall derive the pitch, yaw, and roll components of the vehicle angular rate when the Control Gyro Control Flag is set to 'on'. It shall zero the pitch, yaw, and roll components of the vehicle angular rate when the Control Gyro Control Flag is set to 'off'.

4.1.2.3.1.1.1. Input

The FS shall receive input data from the Control Gyro assemblies.

4.1.2.3.1.1.2. Output

The Control Gyro Processing shall output gyro status and yaw, pitch, and roll vehicle rate data.

4.1.2.3.2. Vehicle Attitude Reference

4.1.2.3.2.1. Horizon Sensor Processing

The Horizon Sensor Processing shall derive vehicle altitude and two axis attitude.

4.1.2.3.2.1.1. Input

The FS shall receive input data from the horizon sensors.

4.1.2.3.2.1.2. Output

The Horizon Sensor Processing shall output horizon sensor status, vehicle altitude data, and two axis attitude data.

4.1.2.3.2.2. Star Scanner Processing

The Star Scanner Processing shall derive three axis vehicle attitude. While the spacecraft is spinning. The processing shall develop a star map for attitude determination.

4.1.2.3.2.2.1. Input

The FS shall receive input data from the star scanner while the vehicle is spinning.

4.1.2.3.2.2.2. Output

The Star Scanner Processing shall output star scanner status and three axis vehicle attitude.

4.1.2.3.3. Vehicle Position/Velocity Reference

4.1.2.3.3.1 Accelerometer Processing

The Accelerometer Processing shall generate vehicle position and velocity vectors.

4.1.2.3.3.1.1 Input

The FS shall receive input data from the accelerometer.

4.1.2.3.3.1.1.2 Output

The Accelerometer Processing shall output vehicle position and velocity vectors.

4.1.2.3.3.2. Vehicle Ephemeris Processing

The Vehicle Ephemeris Processing shall derive the vehicle position and velocity vectors when the Vehicle Ephemeris Control Flag is set to 'on'. It shall zero the vehicle position and velocity vectors when the Vehicle Ephemeris Control Flag is set to 'off'.

4.1.2.3.3.2.1. Input

The Vehicle Ephemeris Processing shall receive inputs from; ground uplinked vehicle ephemerides, vehicle time, and the Accelerometer Processing.

4.1.2.3.3.2.2. Output

The Vehicle Ephemeris Processing shall output vehicle velocity and position vectors.

4.1.2.3.4. Sun Position Update

4.1.2.3.4.1. Sun Sensor Processing

The Sun Sensor Processing shall derive the sun position vector when the Sun Sensor Control Flag is set to 'on'. It shall zero the sun position vector when the Sun Sensor Control Flag is set to 'off'.

4.1.2.3.4.1.1. Input

The Sun Sensor Processing shall receive inputs sun sensor analog and digital signals.

4.1.2.3.4.1.2. Output

The Sun Sensor Processing shall output sun sensor status and sun position vector.

4.1.2.3.4.2. Sun Ephemeris Processing

The Sun Ephemeris Processing shall derive the sun position vector when the Sun Ephemeris Control Flag is set to 'on'. It shall zero the sun position vector when the Sun Ephemeris Control Flag is set to 'off'.

4.1.2.3.4.2.1. Input

The Sun Ephemeris Processing shall receive inputs from ground uplinked sun ephemerides and vehicle time.

4.1.2.3.4.2.2. Output

The Sun Ephemeris Processing shall output the sun position vector.

4.1.2.4. Attitude and Control

4.1.2.4.1. Command Generator

The Command Generator shall generate the commanded vehicle attitude when the Command Generator Control Flag is set to 'on'. The commanded vehicle attitude shall be derived from either the nominal vehicle attitude or from maneuver commands received from RTCs, SPCs, or the Attitude Reference Processing or a combination of them when the Command Generator Control Flag is set to 'on'. The Command Generator shall zero the commanded vehicle attitude when the Command Generator Control Flag is set to 'off'.

4.1.2.4.1.1. Input

The following values shall be the inputs to the Command Generator Processing: Command Generator Control Flag, Command Generator

Command Mailbox, Command Generator Pointer Mailbox, Previous Vehicle Attitude Vector, history parameters.

4.1.2.4.1.2. Output

The following values shall be outputs from the Command Generator Process: Command Generator Module Status Flag, Commanded Vehicle Position Vector, Commanded Vehicle Rate Vector, Commanded Vehicle Acceleration Vector, Command Generator Command Mailbox.

4.1.2.4.1.3. Specified-Profile Maneuver

The Command Generator shall respond to the Specified-Profile Maneuver Command by implementing the specified profile maneuver as given. The maneuver shall be leaked into the Control Law as specified as the Commanded Vehicle Position, Rate, and Acceleration Vectors.

4.1.2.4.2. Vehicle Control Law

The Vehicle Control Law shall generate the commands required to control the MUADEE spacecraft. These shall be comprised of thruster commands and reaction wheel torque commands.

4.1.2.4.3. Attitude Control

The Attitude Control Law shall generate torque commands in response to the commanded acceleration vector, the filtered vehicle rate data, and horizon sensors.

4.1.2.5. Safemode Processing

The Safemode Processing shall run at the 1 Hz rate. The Safemode Data Collection Process shall have modules that run at 10 Hz, 1 Hz, and asynchronously upon demand by computer fault-detection generated interrupts. The safemode processing shall test sensed or calculated values against limits set in the constant data base. A safemode response shall be activated when a tested value exceeds the data base limit for another data base limit consecutive tests. All data base values shall be user configurable.

4.1.2.5.1. EPS Processing

The EPS processing shall maintain an estimated State of Charge for each battery by integrating the current into and out of the battery.

4.1.2.5.2. Safemode Tests

4.1.2.5.2.1. Write Protection Violation

The Write Protection Violation detection shall be implemented as a response to the Write Protection Violation Interrupt. The interrupt response module shall increment the Write Protect Violation Counter whenever a Write Protect Violation Event is detected. The 1 Hz test shall compare this counter to data base limits. When the counter exceeds this limit the safemode test enabled, safemode test activated, safemode response enabled, and safemode response activated values shall be updated; and the MSS Keep Alive process shall be halted.

4.1.2.5.2.2. Processing Rate Timeout

The MSS Scheduler shall verify that each process operates within its allotted time. If a process does not complete within its allotted time then the MSS scheduler shall update the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag, and the MSS Keep Alive Process shall be halted.

4.1.2.5.2.3. 100 Hz vs. 10 Hz sync

The 100 Hz interrupt handler shall verify that the 10 Hz interrupt coincides with the 10th 100 Hz interrupt. If it does not then the sync error counter shall be incremented.

If the sync error counter exceeds a data base limit then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and the MSS Keep Alive process shall be halted.

4.1.2.5.2.4. Interrupt Branch Checksum

The Interrupt Branch Table shall be checksum tested. If the test is failed then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and the keep alive processing shall be terminated.

4.1.2.5.2.6. No MTOC commands for 72 hours

The Command Handler shall maintain the No Uplink Counter - a counter of 10 Hz cycles since the last transmission from the MTOC was received. It shall be incremented by 1 at the start of every Command Handler cycle. It shall be zeroed if an uplink is received.

The No Uplink Counter shall be compared to the data base (72 hours) limit. When this limit is exceeded then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to command the spacecraft into a power positive safemode.

4.1.2.5.2.7. Stored Program List Exhaustion

If the SPC List Exhausted Flag is set then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to command the spacecraft into safemode. The flag can be set in the SPC load or via a real time command.

4.1.2.5.2.8. Sun Protection

The Sun Vector shall be compared to if the angle is less than a data base limit for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to safe the instruments.

4.1.2.5.2.9. No Ephemeris Update

The ephemeris processing shall implement a Ephemeris No Update Counter that counts the number of command cycles since the last ephemeris update.

If the Ephemeris No Update Counter exceeds a data base limit for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to stop the Vehicle Ephemeris Process command the spacecraft into safemode.

4.1.2.5.2.10. SI DPU Toggle

The three Scientific Instrument (SI) Data Processing Units (DPU) shall maintain a toggle bit in the first word of the PIT received by the MUADEE FS from the DPU. The SI DPU PIT processing in the MUADEE FS shall set the Toggle Failure Flag whenever the SI DPU PIT toggle bit fails to toggle; it shall clear the Toggle Failure Flag when the SI DPU PIT toggle bits toggles.

When the Toggle Failure Flag is not zero for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and the SIs and DPUs shall be commanded into a safe state.

4.1.2.5.2.11. Battery State of Charge

The Battery SOC levels shall be compared to a data base first limit. If they fall below this level for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to command the spacecraft into a power positive safemode.

4.1.2.5.2.12 Bus Minimum Voltage

If the Bus Minimum Voltage exceeds data base limits for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and the MSS Keep Alive Process shall be halted.

4.1.2.5.2.13 Total Load Current

If the Total Load Current exceeds data base limits for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to reconfigure the buses.

4.1.2.5.2.14 Total Structure Current

If the Total Structure Current exceeds data base limits for more than data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to reconfigure the buses.

4.1.2.5.2.15 Body Rate Error Check

If the Body Rate Error Flag is set and the gyros are in low mode then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to command the gyros into high mode.

If the Body Rate Error Flag is set and the gyros are in high mode then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to reconfigure to the redundant gyro.

4.1.2.5.2.16 Gyro Reasonableness

If the Control Gyro Data is not within data base limits of the expected values derived from the commanded rate for data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to reconfigure to the redundant gyro pack.

4.1.2.5.2.17 Star Scanner Check

If the Star Scanner Failure Flag is set then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to switch to safemode.

4.1.2.5.2.18 Thruster Current

If the sensed Thruster Current is not within data base limits of the expected values derived from the Thruster Commands for data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to switch the failed thruster to its redundant actuator.

4.1.2.5.2.19 Thruster Pressure

If the sensed Thruster Pressure is not within data base limits of the expected values derived from the Thruster Commands for data base consecutive tests then the safemode test enabled flag, safemode test activated flag, safemode response enabled flag, safemode response activated flag shall be updated; and a macro shall be activated to reconfigure the failed thruster to bypass the pressure loop.

4.1.2.5.3. Safemode Responses

Safemode responses shall be controlled primarily through the use of safemode macros. There are some responses, like the reconfiguration of Thruster Matrices, that can be performed in the software.

4.1.2.5.3.1. Hardware Reconfiguration

The safemode macros shall be able to command the hardware to switch to redundant units.

4.1.2.5.3.2. Software Safemodes

The safemode macros shall be capable of commanding the MUADEE FS into all the levels of software safemode.

4.1.2.6. MSS Performance and Timing

The MSS shall maintain control of the vehicle until such time as a flight computer hardware failure force the MSS to relinquish control to the backup flight computer using the Backup Safemode Software stored in PROM. All of its component modules shall operate within the 100 Hz, 0.1 seconds and 1 second time-slices.

4.1.3. Backup Safemode Software (BSS)

The Backup Safemode Software shall be stored in PROM. It shall use the memory unit of the computer it is called from to store its variable data items.

4.1.3.1. BSS SUN SENSOR Processing

The Sun Sensor Processing shall derive the sun position vector

4.1.3.1.1. Input

The Sun Sensor Processing shall receive inputs sun sensor analog and digital signals.

4.1.3.1.2. Output

The Sun Sensor Processing shall output sun position vector.

4.1.3.2. BSS Command Generator

The Command Generator shall generate the commanded vehicle attitude. The commanded vehicle attitude shall be derived from the sun position vector supplied by the BSS SS Processing.

4.1.3.2.1. Input

The following values shall be the inputs to the BSS Command Generator Processing: Sun Position Vector, Previous Vehicle Attitude Vector, history parameters.

4.1.3.2.2. Output

The following values shall be outputs from the BSS Command Generator Process: Command Generator Module Status Flag, Commanded Vehicle Position Vector, Commanded Vehicle Rate Vector, Commanded Vehicle Acceleration Vector.

4.1.3.3. BSS Vehicle Control Law

The BSS Vehicle Control Law shall calculate the torques necessary to hold the sun vector steady.

4.1.3.4. SUS Initialization

The BSS shall respond to the SUS Initialization Command by activating the SUS.

4.2. Performance and Quality Engineering Requirements

4.2.1. General System Design Requirements

The requirements defined in this section shall be applicable to all CSCs. The intent is to standardize the design to ease the user and ground interface. The requirements defined below shall not apply to the scheduler unless it is specifically identified.

4.2.1.1. Bit-numbering Protocol

Memory shall be referenced as 16-bit words. Within each word the bits shall be numbered from MSB to LSB starting with 0 and continuing to 15 as illustrated below.

msb															lsb
15	14	13	12	11	10	9	8	7	6	5	4	3	2	1	0

4.2.1.2. Control Flags

Each CSC shall be controlled by a flag. When the flag is 'off' the CSC shall transition to a self-initializing state such that when it is reactivated it will be in a known configuration. In the case of processing which stores "past values" of variable parameters, the "past value" buffer shall be set to zero or some other suitable known initial state. If the process such as a command generator is in the middle of an operation, that operation shall be terminated in an orderly fashion if the control flag is set to the "off" state. Each control flag shall be available for telemetry.

4.2.1.3. Self-Initialization

Each CSC shall be self-initializing such that when first activated, it shall always come up in a predetermined state.

4.2.1.4. Mailbox Communication

Each CSC shall have a mailbox interface with the command handler. The mailbox shall reflect the command that is active at any given time. This mailbox shall also reflect a dormant or inactive state. The mailboxes shall be available for telemetry.

4.2.1.5. Accessibility of Constant Data

Data that is defined as constant for a CSC shall be accessible for change by the command handler and shall reside in a constant data base area. Specifically, this shall include control law gains, limiters, error counter thresholds, mounting matrices, scale factors, biases and any other item that may be potentially changed due to a change in physical characteristics of the hardware or space environment. Physical constants (i.e. speed of light) and numerical constants (i.e. pi, pi/2.0) shall reside in a global area accessible by all CSCs.

4.2.1.6. Accessibility of Variable Data

Data that is defined as variable for a CSC shall be accessible for viewing in telemetry as well as for change by the command handler and shall reside in a variable data base area.

4.2.1.7. Definition of Constant Data Base

The constant data base as defined in 4.2.1.5 shall be logically organized by CSC type, and each item shall have the following characteristics.

4.2.1.7.1. Name and Description

Each item shall have a unique name and a clear description.

4.2.1.7.2. Array Indices

Array indices shall be clearly annotated by row, column, unit number, etc. and shall be consistent for all items of the same type.

4.2.1.7.3. Units Specification

Each item shall have a units specification. The metric system shall be used wherever possible.

4.2.1.7.4. Subsystem Traceability

Each item shall be traceable to a subsystem: data management, attitude and pointing control, electrical power, safemode, flight software, or systems engineering. This traceability shall be used to identify responsible parties for a prelaunch data base audit.

4.2.1.7.5. Change History

Each item shall have a change history. For initial build, the value shall be annotated with "Initial". For subsequent changes, the latest Software Change request (SCR) or Software Discrepancy report (SDR) number shall be identified.

4.2.1.7.6. Derivation

Items that are precomputed from a changeable quantity shall show the derivation in the comments field. Multiple items that are derived from the same quantity (e.g. gain, gain/2, .333*gain, etc) shall be organized such that the mutual derivation is obvious and traceable.

4.2.1.8. Definition of Variable Data Base

The variable data base as defined in 4.2.1.6 shall be logically organized by CSC type, and each item shall have the following characteristics.

4.2.1.8.1. Name and Description

Each item shall have a unique name and a clear description.

4.2.1.8.2. Array Indices

Array indices shall be clearly annotated by row, column, unit number, etc. and shall be consistent for all items of the same type.

4.2.1.8.3. Units Specification

Each item shall have a units specification. The metric system shall be used wherever possible.

4.2.1.8.4. Telemetry Available

Inputs to and outputs from a process, as well as selected intermediate values as defined in the algorithm specification, shall be available for telemetry.

4.2.1.9. Numerical and Error Condition Considerations

Flight computers must be able to operate autonomously, handling potential error conditions without ground intervention. While this section does not take the place of a standards and practices document, special attention should be paid to certain conditions that can cause a program to fail. These must be avoided, because if the program fails, vehicle control can be lost.

4.2.1.9.1. Divide Instructions

Divide instructions shall be avoided where possible. Specifically, a divide by a constant should be coded as a multiply by the inverse (1.0/constant). If a divide is required, a logic test shall be performed before the divide to make sure that the denominator is not zero or near zero. The resulting quotient must always be within range of the floating point exponent or fixed point binary scaling.

4.2.1.9.2. Integrator Limiters

Each integrator or counter shall have a limiter, and the limiter shall be resettable in the constant data base. The limiter shall be set to prevent the exponent from reaching the maximum value. They may also be used for control law purposes. Examples include control law integral paths, error counters, timeout counters, etc.

4.2.1.9.3. Error Condition Handling

Processing error conditions shall be handled by the process detecting the error. A telemetry monitor shall be made available to notify the ground segment that an error has occurred. Some errors may actually be safemode responses or can be avoided through proper setting of the data base. These error types shall not be handled by the process.

Mars Upper Atmosphere Dynamics Energetics, and Evolution Mission

Spacecraft Radiation Environmental Estimates

Al Howard

J.A. Howard Jr.

14.2 RADIATION ENVIRONMENTAL ESTIMATES

This document defines the radiation environmental estimates for the MUADEE spacecraft. These requirements are established to serve as a basis for deriving the radiation environmental design and test requirements for the spacecraft system and each assembly.

14.2.1 Natural Charged Particle Radiation Environment

The natural charged particle radiation environment for the MUADEE mission described in this section is derived from the Mars Geoscience Surveyor Environmental Estimates. The Mars Geoscience Surveyor is expected scheduled launch in late 1996 while the MUADEE mission's earliest launch opportunity is around December 1998. This means that the MUADEE mission will begin around the start of active sun.

14.2.2 Electron and Proton Environment.

Figures 14.1 to 14.4 show the estimated electron and proton fluxes and fleuences. The start of active sun is 2.57 years.

14.2.3 Total Ionizing Dose (ID) Radiation Environment.

Figures 14.5 to 14.7 show the Ionizing dose for various thickness spherical shells vs. mission time. No design margins are included. Figure 14.8 shows the total dose vs. time. Figure 14.9 shows the typical and peak total dose rate.

14.2.4 Single Event Effects (SEE).

In addition to protons and electrons, heavy ions occur that can cause SEEs. Sources for heavy ions include galactic cosmic rays, ions accelerated in the interplanetary medium, and particles from solar flares. The heavy ion flux with linear energy transfer (LET) greater than the indicated value is given by the Heinrich curves of figure 14-10. These values are appropriate for the entire mission.

Figure 14-1

ELECTRON FLUENCE FOR MUADEE MISSION

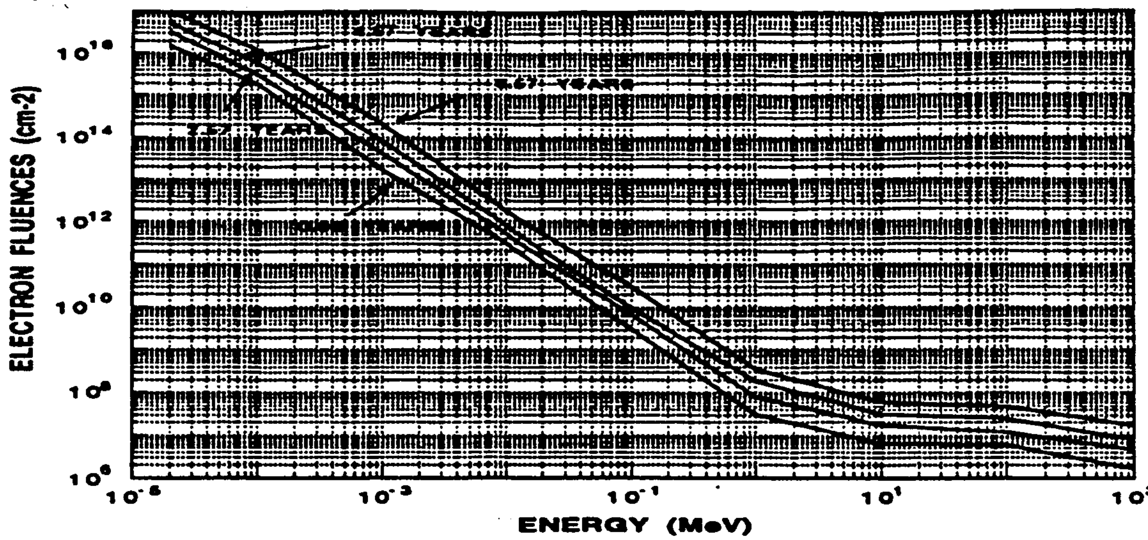


Figure 14-2

MUADEE PROTON FLUENCES

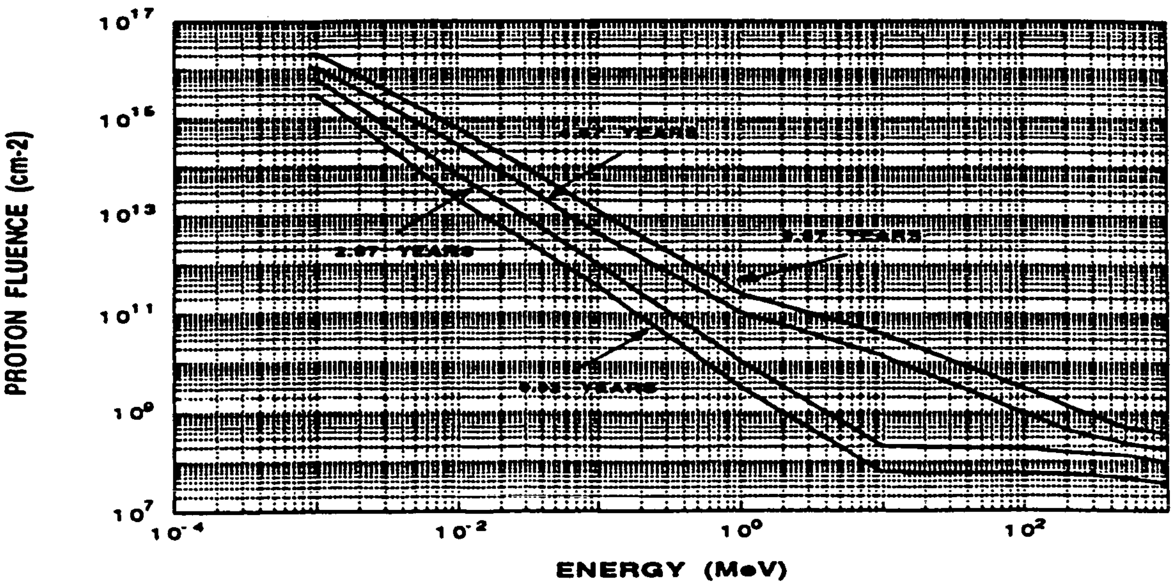


Figure 14-3

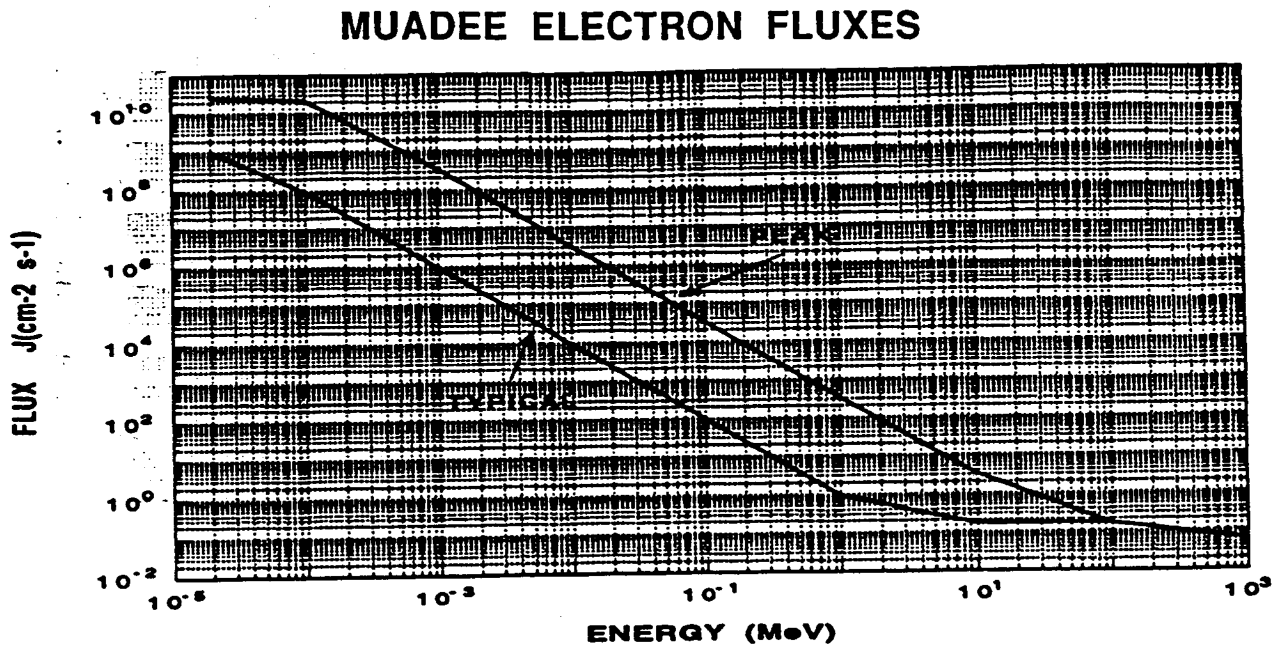


Figure 14-4

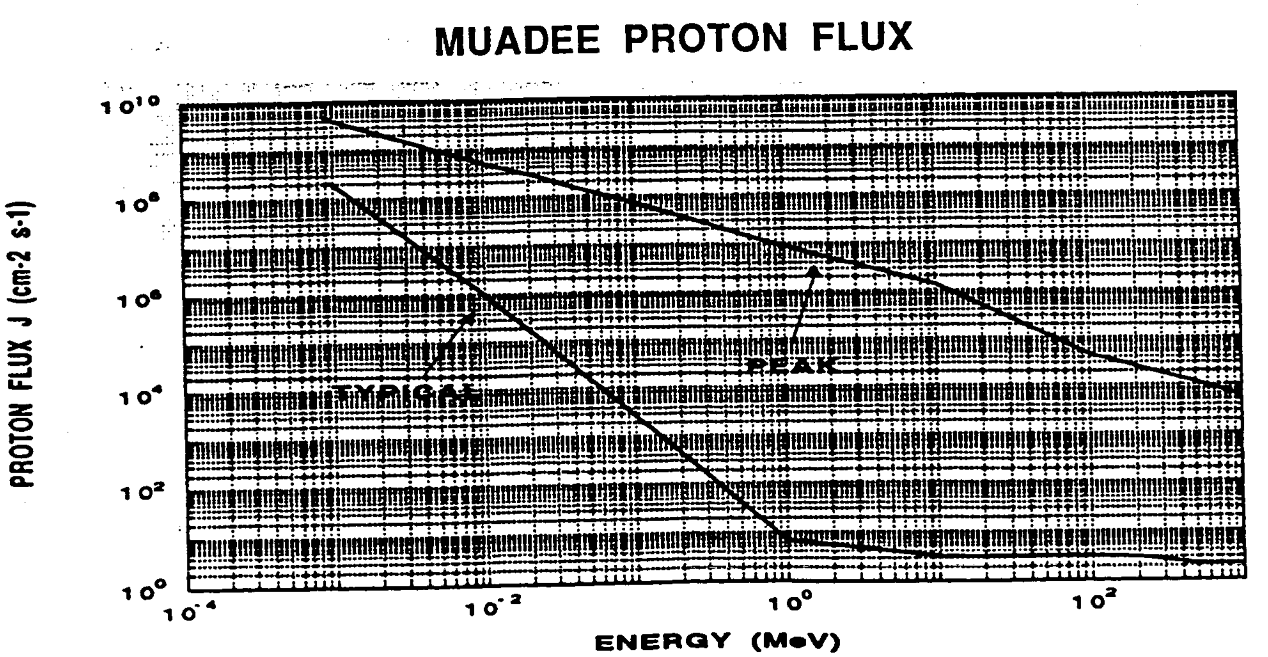


Figure 14-5

MUADEE ELECTRON DOSE

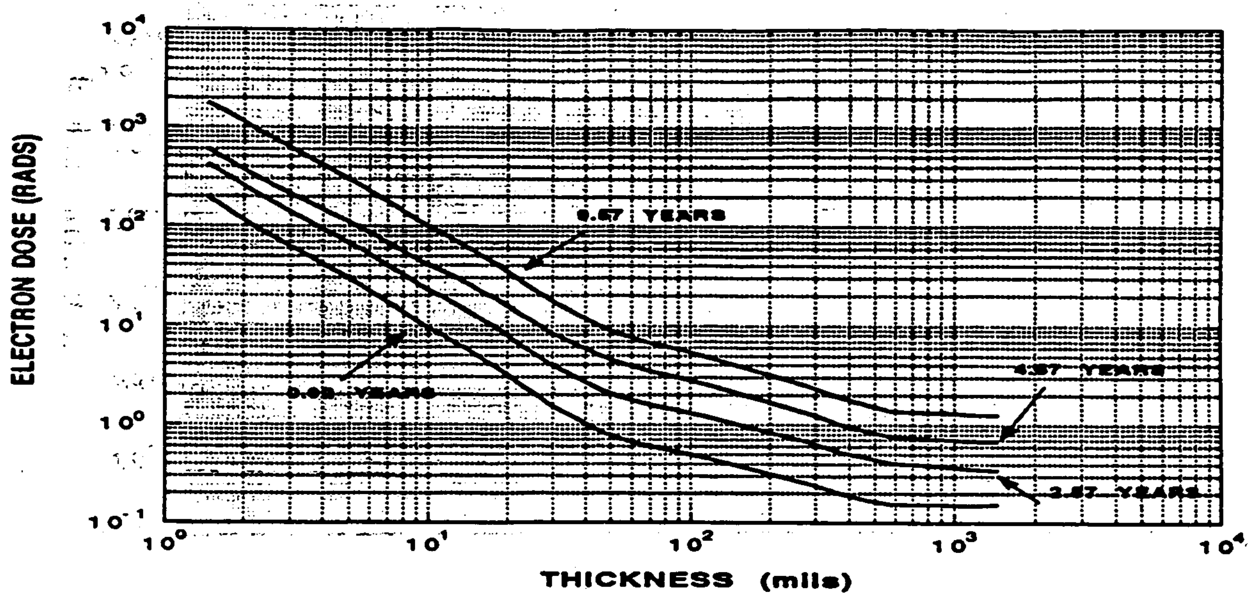


Figure 14-6

MUADEE PROTON DOSE

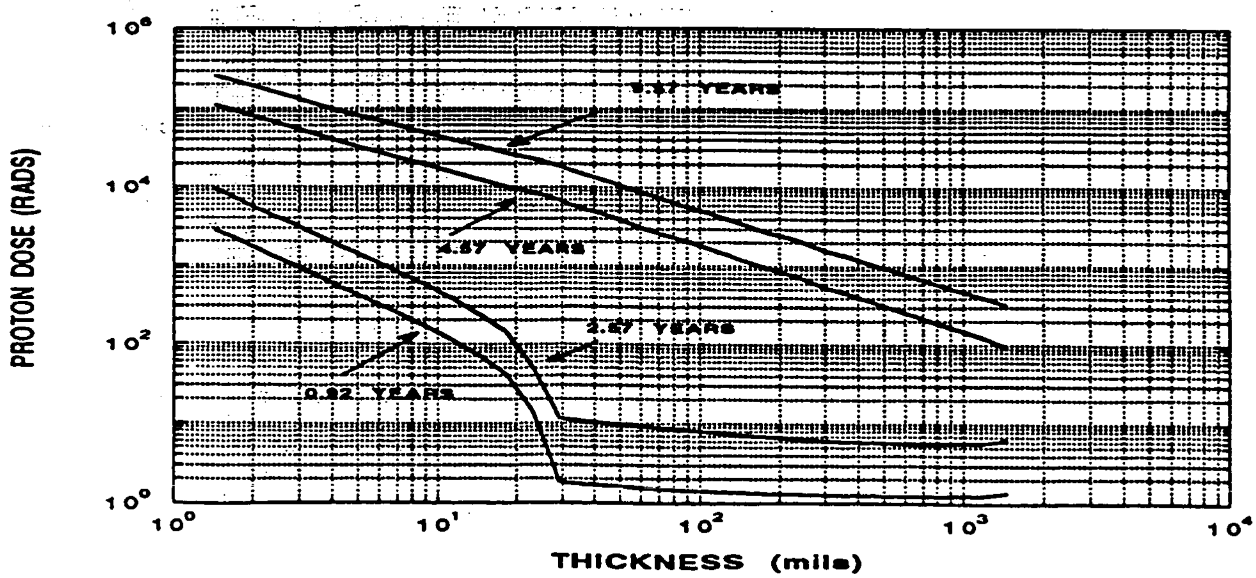


Figure 14-7

MUADEE TOTAL RADIATION DOSE

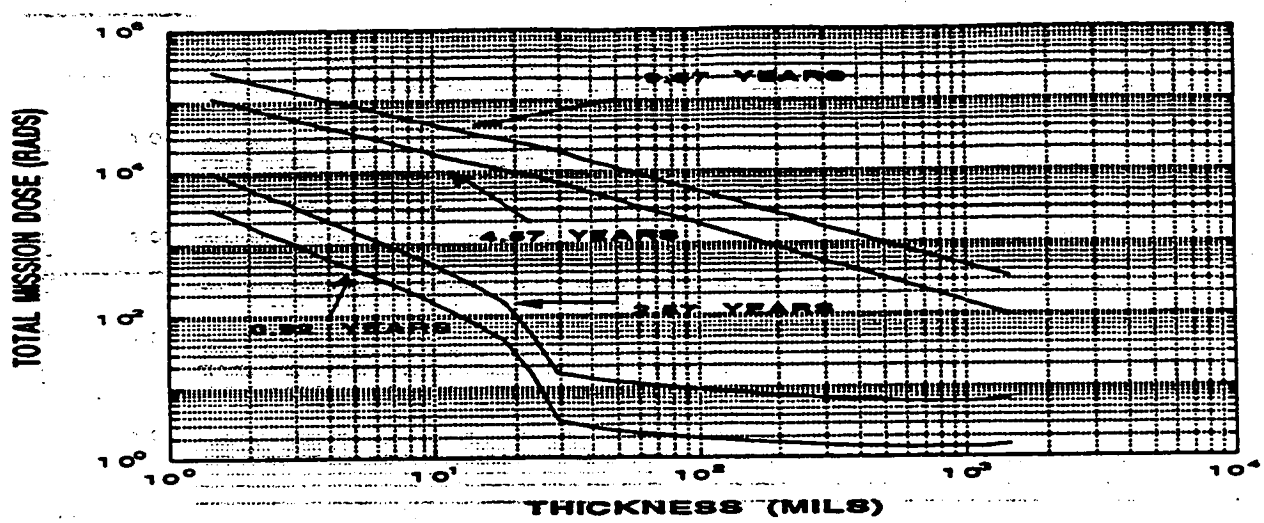


Figure 14-8

MUADEE TOTAL DOSE AS A FUNCTION OF TIME

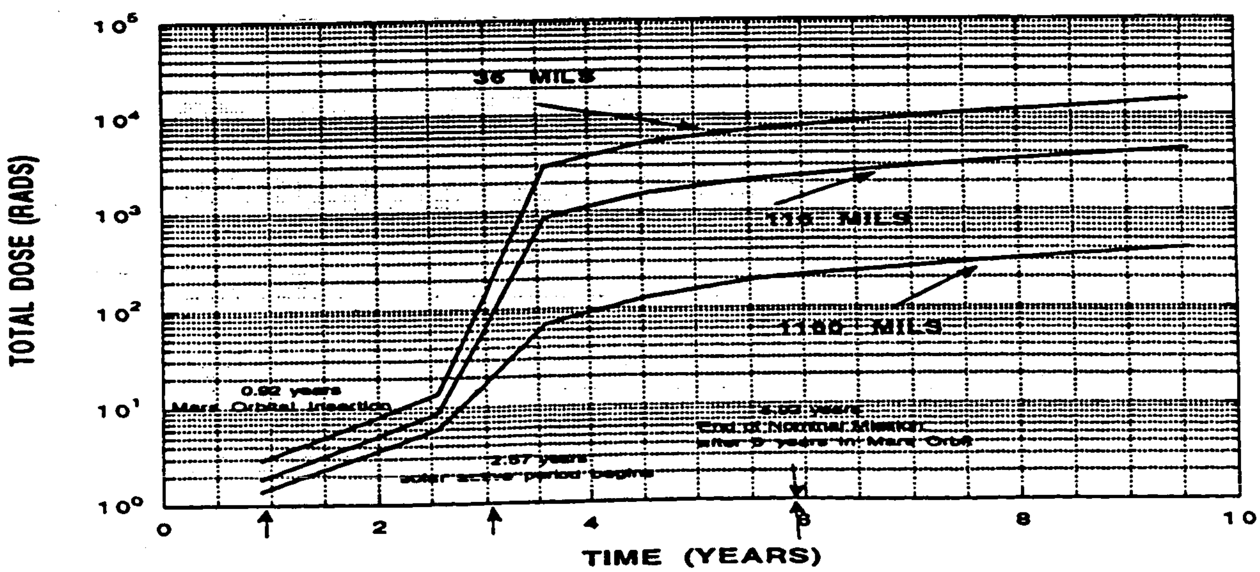


Figure 14-9

MUADEE TOTAL DOSE RATE

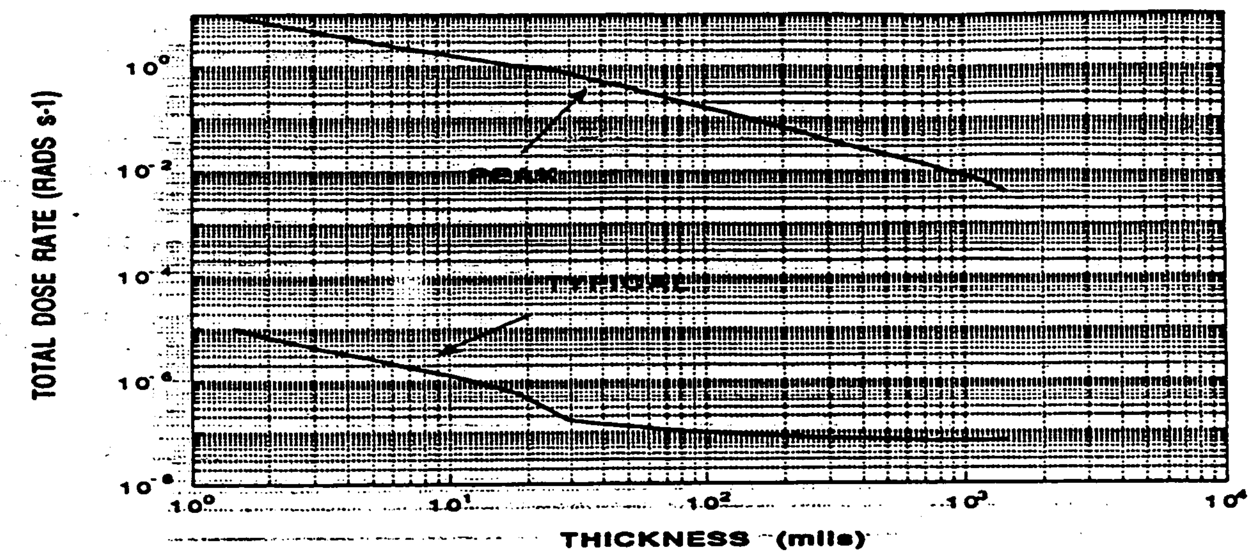
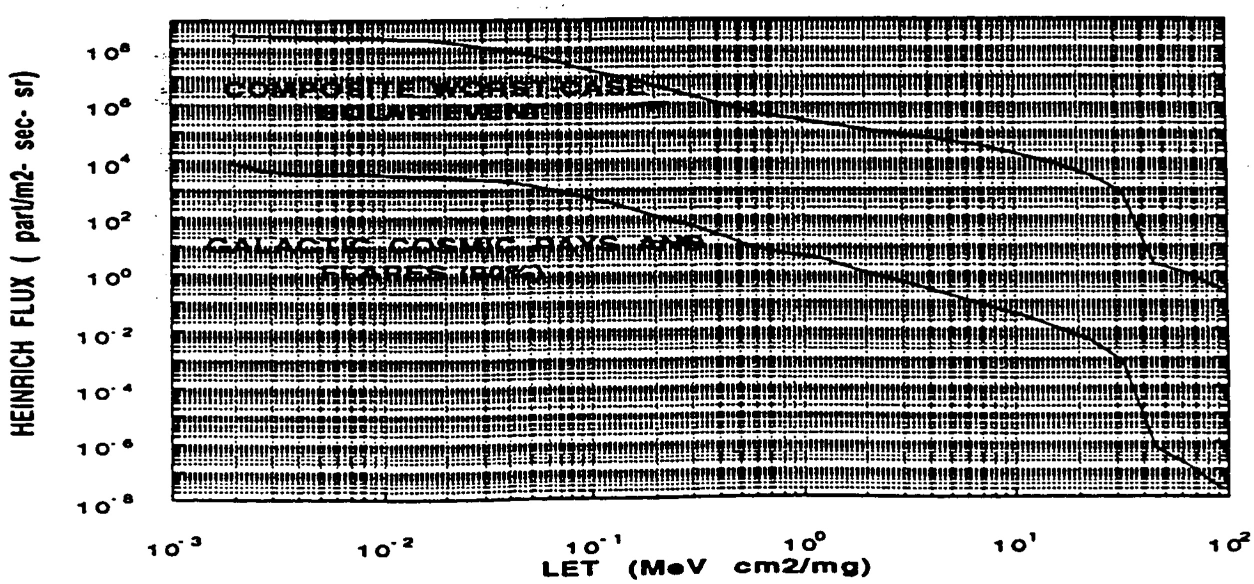


Figure 14-10

MUADEE
HEINRICH FLUXES FOR 25 mil SPHERICAL SHELL



3/15/94

PRODUCT ASSURANCE PLAN

FOR

THE MUADEE SPACECRAFT PROGRAM

Prepared by: K Sagara

K. Sagara

TABLE OF CONTENTS

TABLE OF CONTENTS CONTINUED:

PRODUCT ASSURANCE PLAN

1.0 Scope

This document describes the Product Assurance Program and Inspection System LMSC will use on the MUADEE spacecraft program to ensure conformance with contract requirements.

2.0 Applicability

This plan shall be applied to all products and services provided by the MUADEE program. All work performed within MUADEE program areas shall be governed by this plan.

3.0 Contractual Intent

The intent of this Product Assurance Plan is to describe LMSC/SSD product assurance requirements for the MUADEE program. Requirements are tailored versions of LMSC/SSD policies, procedures, and quality procedures developed for Department of Defense (DOD) projects. In some cases, special procedures may need to be prepared by MUADEE program personnel for unique tasks. For these situations, the special procedures shall take precedence.

4.0 Summary

This product assurance program provides for logical and economical assignment of functions that contribute to effective management toward the quality of MUADEE spacecraft and/or ground support hardware/software products.

The MUADEE program manager maintains responsibility for overall system quality and adherence to contract requirements. The MUADEE product assurance program management representative (PAPMR) is responsible for verification of conformance quality.

Objective evidence of quality conformance is provided to the designated University of Michigan representative to provide the basis for product acceptance.

5.0 Relation To Other Contract Requirements

The product assurance program requirements described in this plan shall be implemented by the MUADEE PAPMR to satisfy detailed requirements established by the MUADEE contract and statement of work.

6.0 Applicable Documents

The following documents of the issue in effect on the date of invitation for bids form a part of the specification to the extent specified herein:

NHB 5300.4(1B)	Quality Program Provisions for Aeronautical and Space System Contractors
NHB 5300.4 (1C)	Inspection Provisions for Aeronautical and Space System Materials, Parts, Components, and Services

MIL-I-45208 Inspection System Requirements

7.0 Use of MIL-I-45208

The requirements of MIL-I-45208 are used for general procurement of non complex hardware, materials, and services. If procurements for the MUADEE program are made of those items from suppliers not approved to MIL-I-45208, then NHB 5300.4(1C) applies.

8.0 Quality Program Management

8.1 Organization

The MUADEE PAPMR, or his designee, represents the SSD Product Assurance organization with signature authority to assess, evaluate, and ensure prompt corrective action on any element of the quality program that may affect MUADEE program hardware/software quality as implemented. The MUADEE PAPMR coordinates and implements all MUADEE system effectiveness requirements, including safety, reliability, electrical, electronic and electromechanical (EEE) parts, material and processes, and product assurance.

8.2 Initial Quality Planning

Planning for quality begins with the initiation of the MUADEE program and is integrated into the Product Development Teams (PDT's). The PDT's are responsible for design of spacecraft components, procurement, manufacturing, and testing activities. The PAPMR or his designee is a member of PDT's.

Processes requiring special controls, test equipment, facilities, tools, fixture, instructions, and personnel training are identified, and satisfactory control is implemented in a timely manner under the cognizance of the MUADEE PAPMR or his designee.

Quality assurance personnel participate in design reviews to ensure that producibility, repeatability, inspectability, and related quality considerations are incorporated in the design.

8.3 Work Instructions

All work affecting quality is prescribed in clear and completed documented instructions appropriate to the circumstances.

The MUADEE PAPMR is responsible for the review and approval of flight hardware documents that are used to procure, fabricate, test and inspect MUADEE flight hardware to ensure adequacy of the work instructions with respect to the inclusion of quality requirements

Work instructions include:

- Design — Engineering drawings and specifications - review before, during, and after release.

- Procurement — Purchase request, purchase orders, subcontracts, and statement of work - review and approve.

- Manufacturing — Shop work authorizing documents and manufacturing process standards - review and approve as required by the MUADEE program office.

- Test — Test plans, test procedures, and operations orders - review and approve as required by the MUADEE program office.

- Inspection — Quality operations instructions and product assurance standards - prepare and implement as required by the MUADEE program office.

8.4 Records

Job Package Authorizations (JPA's), engineering logs, operations records, and log books used for shop work authorizing documents (SWAD), fabrication, processing, and assembly of the deliverable hardware will be maintained by the MUADEE data center. These documents shall be maintained and made available for review by the University of Michigan, and copies furnished, as required, to the university. These documents indicate the operations performed, inspections performed, the acceptability of the work operation, and actions taken in connection with non conformance through reference to discrepancy logs (D-logs).

8.5 Corrective Action

D-logs document each product non conformance and provide a record of disposition. D-logs prove for failure analysis and corrective action as deemed essential by the MUADEE program office. Historical D-log records are maintained by the MUADEE data center. Discrepancies are signed off by the MUADEE PAPMR.

8.6 Drawing, Documentation, and Changes

Essential elements of the drawing control system are documented in the MUADEE Management Plan.

8.7 Measuring and Testing Equipment

Suitable inspection, measuring, and test devices required to prove the quality performance and that dimensions are maintained are under the direction of the PDT as part of the quality program. Calibration or re calibration of inspection, testing, and measuring equipment is certified to standards traceable to the National Bureau of Standards.

The MUADEE PAPMR/inspector audits measuring and test devices to ensure that they are current in calibration and repair. Calibration of inspection equipment is in accordance with MIL-STD-45662.

8.8 Production Tooling Used for Inspection

Suitability of production tooling used as a medium for inspection is determined by the appropriate MUADEE program PDT. Production tooling may be used for inspection whenever the level of product acceptability can be determined more efficiently and economically by this means. The PAPMR or his designee inspects and approves new, reworked, or modified tools or tool details used for inspection to acceptance criteria specified on the tool order or engineering design drawing. Tool proofing is performed as required.

Tool inspection log books reflecting the comprehensive tool history are prepared and maintained for tools when required. Each tool inspection log book is retained with the associated tool by the MUADEE program.

8.9 Use of Contractor Inspection Equipment

The University of Michigan may use any LMSC inspection equipment to verify inspection accuracy, product quality, and conformance to contract requirements.

9.0 Control of Purchases

9.1 Responsibility

The MUADEE program office maintains a system for ensuring that all supplies and services in support of the MUADEE program conform to contract requirements. Purchases of flight items are made only from suppliers listed in our directory of approved suppliers (DAS).

9.2 Purchase Request Review

Purchase requests (PR's) and purchase orders (PO's) are reviewed and approved by the MUADEE PAPMR, before buyer action, to ensue inclusion of applicable supplier quality requirements, to ensure completeness and adequacy of the PR document as it is required to conform to MUADEE program requirements, and to ensure that the supplier is in the DAS.

9.3 Subcontract Document Review

The MUADEE PAPMR reviews all subcontractor proposals to ensure that appropriate MUADEE quality requirements are flowed down to the subcontractor.

9.4 Supplier Source Inspection

The MUADEE PAPMR is responsible for source inspection of designated items to provide assurance that a supplier's techniques, tooling, and methods produce products in conformance with MUADEE requirements. The items to be source inspected are determined by the PDT. The purpose of such inspection is to minimize the possibility that a supplier may produce and deliver a production lot of material that would be subject to rejection upon presentation for acceptance. All final acceptance tests on subcontract hardware are witnessed by the MUADEE PAPMR or his designee, who could be the subcontractor product assurance representative.

9.5 Supplier Audits

Auditing of MUADEE subcontractor shall be performed by the MUADEE PAPMR as required.

10.0 Manufacturing Control

10.1 Materials and Product Control

Supplier's materials and products shall be subjected to inspection by the MUADEE PAPMR as required upon receipt into MUADEE program areas to ensure conformance to MUADEE program requirements.

10.2 Receiving Acceptance Tests

Final acceptance testing of procured hardware (subcontract or PO) is performed at the supplier's facility using procedures approved by the MUADEE PAPMR and forms the basis for hardware acceptance. Incoming inspection occurs in the MUADEE designated program area. Any testing performed at MUADEE incoming inspection is performed at the direction of the MUADEE PAPMR.

10.3 Mechanical, Identification, and Damage Receiving Inspection

All productive supplies are inspected at the MUADEE receiving area for identification, shipping damage, and evidence of inspection as may be required by the procurement document.

10.4 Raw Material Control

Raw, wrought, or bulk materials procured for flight use will have test reports, certificates of conformance, or other documents as needed to show objective evidence of quality conformance. These documents are approved by the MUADEE PAPMR. Requirements are invoked on suppliers to ensure control of raw materials used in production of items they supply. Assurance of supplier conformance to specified controls is maintained by the MUADEE PAPMR using system quality audits.

10.5 Inspection Planning

Inspection instructions are prepared by the MUADEE PAPMR to direct inspectors in determine hardware conformance to design requirements. Inspection instructions are included on the SWAD or other documents such as receiving and inspection instruction, and test procedures.

10.6 Quality Engineering

The MUADEE PAPMR provides quality engineering and inspection support to manufacturing and engineering as required. Changes to SWAD's are approved to ensure that changes have no adverse effect on work or inspection operations. SWAD change approval by the MUADEE PAPMR is noted by stamping, signing, and dating the SWAD.

10.7 Completed Item Inspection and Testing

10.7.1 Configuration Verification

Manufacturing and test SWADs reflect final configuration reconciliation by showing the engineering configuration at time of SWAD release plus subsequent changes. This configuration is verified by production and inspection as also being the manufactured configuration. Acceptance of all in-process operation is indicated on the SWAD by production and inspection stamps and date.

10.7.2 Inspection, Test, and Test Surveillance

Monitoring of testing performed by manufacturing and engineering organizations follows procedures approved by the PDT. This ensures compatibility of procedures to product assurance provisions and design documentation, inclusion of acceptance criteria, clarity of presentation, and adherence to MUADEE program format standards.

Modifications after final inspection or test are re inspected and retested to the completed item inspection and test procedures to the extent necessary to verify performance of the article.

Inspection and test data are recorded and maintained by the MUADEE data center.

10.8 Packaging, Handling, and Storage

Materials and supplies manufactured or purchased in support of the MUADEE program are packaged, handled, and stored as required by engineering data related to the specific material or supply. The MUADEE PDT establishes preservation and packaging requirements.

The MUADEE PAPMR monitors packaging, handling, and preservation to protect product quality and prevent product damage, loss, deterioration, degradation, or substitution.

11.0 Delivery

The MUADEE PAPMR is responsible for compiling documentation required for readiness reviews in support of product delivery. The MUADEE PAPMR inspects preparation for outgoing shipment, checks shipping documentation, and witnesses loading to ensure compliance with contract requirements, engineering data, and carrier regulations as applicable at the time of DD250.

12.0 Non conforming Material

A closed-loop system of material review is maintained for dispositioning non conforming material. Minor discrepancies may be corrected on the floor with no accompanying documentation. Discrepancy logs shall be used for "repair" or "use-as-is" dispositions. Discrepancies are dispositioned by the MUADEE program material review board (MRB). This MRB comprises the PDT leader, the MUADEE PAPMR, and a University of Michigan representative. A system for control of non conforming material within the MUADEE program area is maintained by the MUADEE PAPMR., including procedures for identification, segregation, and disposition.

13.0 Statistical Quality Control

Statistical quality control is not used except as directed by the MUADEE program office.

14.0 Indication of inspection Status

A system is maintained to identify product status. These procedures prescribe the method for showing in-process acceptance of a product, including special processing, final acceptance of product, product rejection, voiding previous status and re identifying, and for audit use and control of inspection stamps.

15.0 Coordinated Government and Contractor Actions

15.1 Government and Customer Inspection at Subcontractor or Vendor Facilities

The MUADEE program specifies government source inspection (GSI) on purchased supplies only when requested or authorized by NASA or University of Michigan representatives.

15.2 Government Property

Inspection and reporting of damaged NASA or University of Michigan furnished material are performed by the MUADEE PAPMR. Discrepant NASA or Michigan furnished material shall not be dispositioned or modified in any way without prior customer approval.

16.0 Software Quality Assurance

The PAPMR monitors the software development process to ensure that configuration management is performed. He ensures that computer based software discrepancies reports (SDR's) and change requests (SCR's) are properly logged and dispositioned, that all requirements are implemented, and that controlled releases are used for formal verification testing. He maintains a tape library of tested software releases, with a remote site backup for disaster recovery. He works with the configuration management specialist to ensure that all SDR's, SCR's, software releases, and documentation are indexed, archived, and configuration controlled.

The formal verification and validation testing of software products is performed by the Software Test and Integration group. This group develops test plans and procedures to test each computer software configuration item (CSCI). The plans and procedures establish test criteria; defines the test inputs and database applicable to each test item; and describes step-by-step procedures for each test activity and demonstration required to verify the performance, correctness, and satisfaction of software requirements. The test and integration group conducts software verification and hardware/software validation tests. They evaluate all results and prepare related test reports. They schedule tests, ensure the presence of all participants at the tests, and maintain the computer based Software Development Library containing the software under test until it is turned over to the PAPMR.

Major software modules to be developed are divided into flight, system test, and ground segment CSCI's. Commercial off-the-shelf or government furnished software is used to the extent possible to minimize cost and development time.

INTERDEPARTMENTAL COMMUNICATION

To: P. Williams **Org.:** 6N-02 **Bldg.:** 107 **Date:** 4 Mar. 1994

From: D. H. Utter **Org.:** 74-16 **Bldg.:** 580 **Phone:** x26275

Subject: MUADEE Six DOF Simulation

Summary:

This report describes the six degree-of-freedom Mars simulation for attitude control design and analysis of the MUADEE mission phases and control modes. Several examples of pointing error times histories are shown.

Discussion:

The simulation used for the MUADEE analysis was rapidly reconfigured for Mars orbit from a simulation program developed for analysis of earth orbiting space vehicles. It is a six degree-of-freedom, time domain simulation written in MATRIXx superbuild blocks and user code blocks hosted on a Sun Sparc 2 workstation.

The simulation was used to evaluate the MUADEE Attitude Control Subsystem (ACS) pointing accuracy performance for several mission orbits and control modes. Examples are shown later in this document.

This discussion is divided into two parts: the first describes the simulation model, while the second discusses the ACS pointing results for both elliptical and circular orbits.

<u>Simulation Model</u>

The simulation model is organized in a hierarchical array of MATRIXx superblocks. The top-level superblock is named "A MARS Sim" and is shown in Figure 1. It contains five major sub-blocks: an Initial Conditions superblock, a Mode Logic superblock, an Environment superblock, a spacecraft Vehicle Dynamics superblock, and a Vehicle Subsystems superblock. Each of these major superblocks will be described below. The input to the top-level superblock is a time vector. The output is currently an array of 64 variables for each input time point, but it can be set up with any length desired.

The Initial Conditions superblock sets up the initial conditions for the Mars orbiting spacecraft position, velocity, attitude, angular rate, and weight.

The Mode Logic superblock determines the simulation mode sequence logic and starting mode.

The Environment superblock calculates the timebase model, a Mars atmosphere model, and a wind model. The atmosphere model is a linearized lookup of atmospheric density vs. altitude for altitudes between 100 and 345 km.

The Vehicle Dynamics model contains all the position/velocity and attitude quaternion/angular rate integrations. It is organized into sub-blocks for orbit dynamics and gravity, aerodynamics, summation of forces and moments, rotational kinematics, mass/inertia properties, reference coordinate frames, propulsion model, and reaction wheel model.

The orbit dynamics block is a User Code block for the Mars orbit dynamics with Mars gravity model, which has specified equatorial and polar radii and J2 through J5 terms.

The aerodynamics model computes aeordynamic forces and moments in body coordinates based on coefficients as functions of both angle-of-attack and slideslip angle.

The rotational kinematics model computes angular accelerations and quaternion rates, based on input torques, inertia properties, and reaction wheel angular momentum, and integrates the derivatives. The resulting angular rate and attitude quaternion are used in many other blocks in the simulation.

The mass/inertia properties block used for this analysis contains constant values. (In future versions, propellant mass will vary with delta V burns and reaction control thruster (RCT) usage.) The current mass is 2200 lbs (1000 kg), with spin and transverse axis moments of inertia of 441 slug-ft*2 (600 kg-m*2) and 331 slug-ft*2 (450 kg-m*2).

Several coordinate frames are modelled. They are as follows:

ECI	earth (Mars) center inertial	x,y thru equator, z thru North pole
LVLH	local vertical, local horiz	x forward, y orbit perp, z nadir
Limb	forward limb pointing	x @ fwd limb, y orbit perp, z down
Body	body fixed	x spin axis, y,z transverse

The propulsion model contains linearly proportional valves for the reaction control thrusters (RCTs). The current RCTs are 0.5 lb (2.2 N) on moment arms of 3.6 ft (1.1 m). There are two RCTs per axis.

The reaction wheel model is currently one reaction/momentum wheel with initial spin rate of 6000 RPM and angular momentum of 60 ft-lb-sec.

The Vehicle Subsystem model currently assumes perfect attitude and rate sensors. It consists of an attitude controller and a reaction wheel controller.

The attitude controller is used in two configurations, a three-axis mode and a spin stabilized mode. For the three-axis mode, the roll command is sent to the reaction wheel, while the pitch and yaw commands are sent to the reaction control thrusters. For the spin-stabilized mode, the roll attitude error and pitch and yaw channels are disconnected, and only the roll rate error is sent to the reaction wheel while the RCT commands are zeroed. (A more sophistocated controller will be developed later to use the RCTs to keep the spin axis aligned along the orbit perpendicular.)

The reaction wheel model applies torque to the reaction wheel depending on the roll error for the three-axis mode. For the spin stabilized mode, the wheel is commanded to zero RPM and the vehicle is commanded to 3 RPM.

ACS Pointing Results

In the three-axis mode, pointing error is defined as the RSS of the three Euler angles, yaw, pitch, and roll, of the body axis frame relative to the Limb coordinate frame. In the spin mode, pointing error is defined as the RSS of the two Euler angles, azimuth and elevation, of the vehicle x-axis relative to the orbit perpendicular of the Limb coordinate frame.

The pointing errors depend primarily on the frequency content of the disturbances and the bandwidth and type of attitude controller. For the three-axis stabilized control mode, the attitude controller was a proportional-integral-derivative (PID) type with a bandwidth frequency of 0.05 Hz.

The dominant disturbance is considered to be the aerodynamic torques caused by the high-gain antenna mounted on the "top" of the spacecraft. The spacecraft normally flies with the "top" or spin axis to the left of the velocity vector or along the orbit plane perpendicular, and one side of the cylinder points toward the forward limb. The spacecraft is either 3-axis stabilized or rotates about the spin axis. In this configuration, a yaw torque is generated by the high-gain antenna. The amount of torque depends on the dynamic pressure.

Typical pointing errors for one circular orbit at 250 km altitude in the 3-axis stabilized mode are shown in Figure 2. The pointing error, neglecting the startup transients, is approximately $\pm$ 0.05 degrees peak to peak. Conservatively, 0.1 is taken as the 3-sigma statistical error.

Typical pointing errors for one elliptical orbit of altitude 2500 by 130 km in the 3-axis stabilized mode are shown in Figure 3. The pointing error is taken to be 0.26 degrees, 3-sigma, as the worst case error.

Typical pointing errors for the circular orbit at 250 km in the spin mode are shown in Figure 4. In this case there is negligible aero disturbance and no reaction control thrust. The vehicle merely spins with a minimum cross-product of inertia giving a wobble angle. It can be seen that the pointing error, which is that of the vehicle x body axis pointing relative to the inertial orbit plane perpendicular direction, is on the order of $\pm$ 0.25 degrees.

Devin Utter
SSD Spacecraft Control Products, O/74-16

Discrete SuperBlock	Sampling Interval	First Sample	Ext.Inputs	Ext.Outputs	Enable
A MARS Sim	0.0200	0.	0	64	Parent

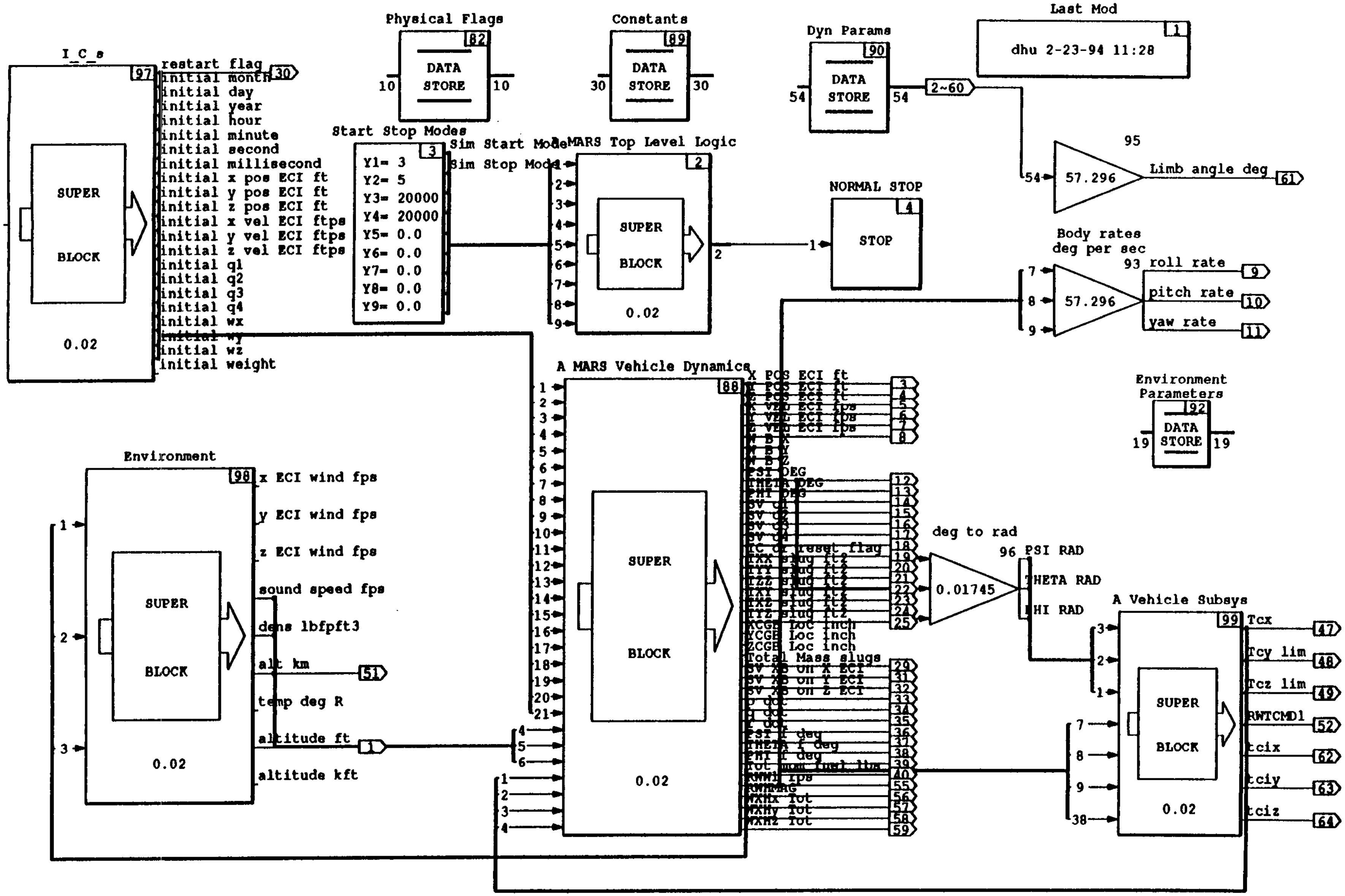


FIGURE 1. TOP-LEVEL SUPERBLOCK

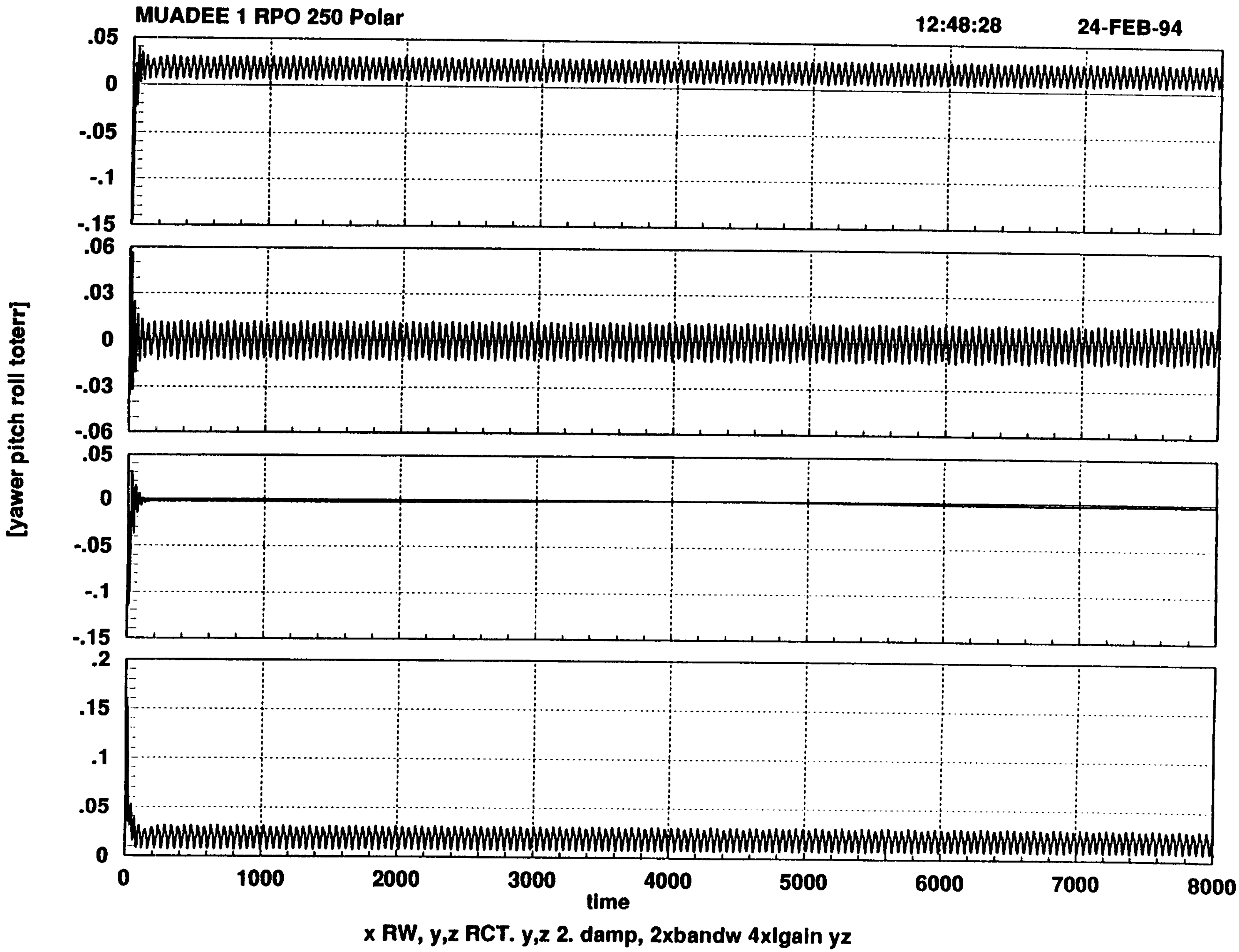


FIGURE 2. POINTING ERRORS, CIRCULAR ORBIT, 3-AXIS MODE

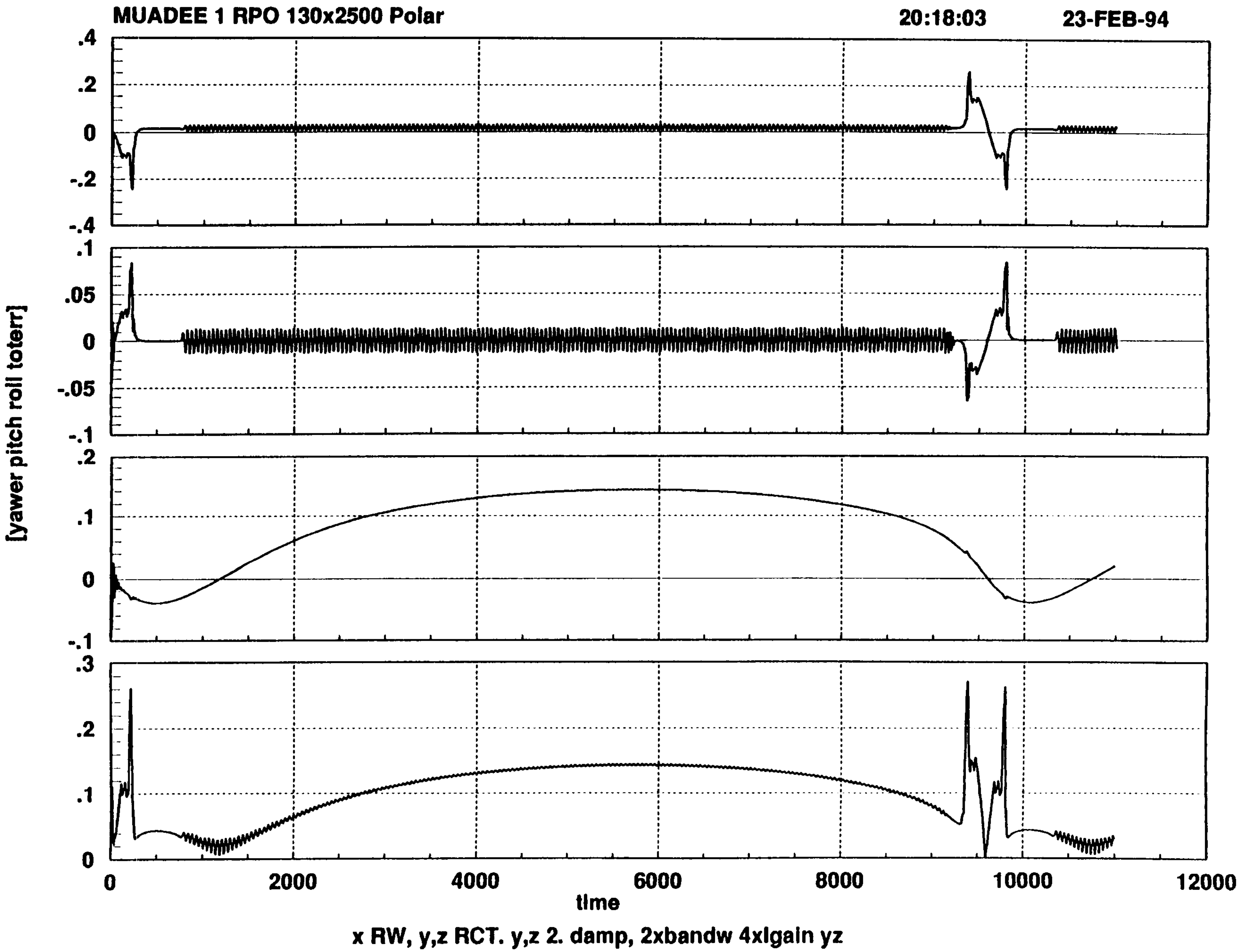


FIGURE 3. POINTING ERRORS, ELLIPTICAL ORBIT, 3-AXIS MODE

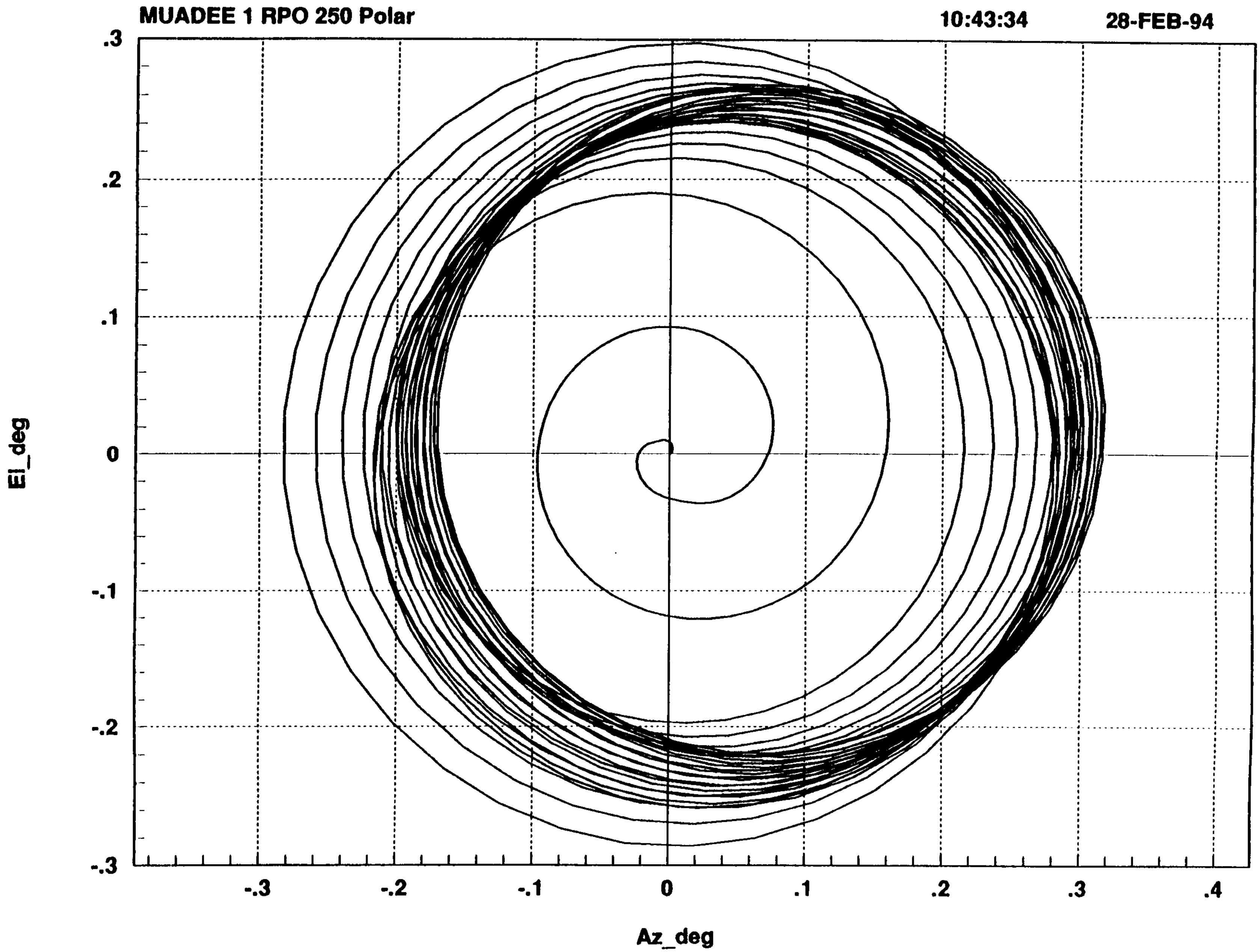


FIGURE 4. POINTING ERRORS, CIRCULAR ORBIT, SPIN MODE

INTERDEPARTMENTAL COMMUNICATION

To:	P. Williams	Orgn: 77-40	Bldg: 107	Date: 2-25-94
From:	M. Hilton	Orgn: 77-40	Bldg: 580	Ext: 60258

Subject: MUADEE Structural Model

The purpose of this IDC is to document the stowed finite element model developed in support of the MUADEE contract and the assumptions and techniques used.

The model was created using the I-DEAS finite element pre-processor. It consists of 720 nodes and 1010 elements. The structure included in the model consists of the bus structure, the booster adapter cylinder and the HGA subassembly. The following components are modeled as lumped masses: payload instrument packages, wet propellant tanks, reaction wheels, HGA, PSDU, battery, Flight computer and VMAG boom and canister. The solar cell mass is distributed over the solar cell panels located on the outer cylinder panels and upper bus panel. The remaining mass consisting of smaller electronics boxes, thermal control equipment and wire harnesses is distributed evenly over the equipment panels -- the radial, upper and lower honeycomb panels.

The bus is modeled using beam element to represent the aluminum frame and plate elements to represent the aluminum shells and honeycomb panels. The honeycomb panels are modeled in the laminate modeling module within I-DEAS which creates an orthotropic material property and allows facesheet stresses to be calculated and graphically displayed in the post-processing module.

The booster adapter is modeled with 72 .063" plate elements. The 24 nodes at the Booster interface are pinned representing a marmon clamp interface.

The conical HGA boom platform is represented by 72 .063" plate elements. The Gr/E HGA boom is modeled with a beam element. The HGA dish, gimbals and electronics are, as mentioned above, modeled as a lumped mass at the upper end of the beam.

The model is correlated to the latest mass property statement as detailed below.

Mass = 897 KG
$I_{xx} = 423$ Kg-m^2
$I_{yy} = 380$ Kg-m^2
$I_{zz} = 260$ Kg-m^2

As a check on the validity of the model, static reactions forces were summed and found to equal the total applied load. Additionally, no spurious restraints were detected in the output. These two results indicate that the model is properly restrained and the results are credible.

The The final report charts relevant to the model are attached for completeness.

MUADEE

ANALYSIS ASSUMPTIONS AND METHODOLOGY

- DISCRETE LUMPED MASSES TO REPRESENT:
  - INSTRUMENT PACKAGES
  - PROPELLANT TANKS AND FUEL
  - REACTION WHEEL
  - HGA
  - PSDU, BATTERY, COMPUTER
  - VMAG BOOM AND CANISTER
- S/A CELL MASS EVENLY DISTRIBUTED OVER SOLAR ARRAY SUBSTRATE PANELS
- REMAINDER OF MASS DISTRIBUTED EVENLY ON RADIAL PANELS AND UPPER AND LOWER BUS PANELS
- HGA MODELED AS A RIGID MASS MOUNTED ON A FLEXIBLE BOOM AND PLATFORM

MUADEE

MUADEE STRUCTURAL PERFORMANCE SUMMARY

- DESIGN MEETS ALL STRUCTURAL REQUIREMENTS WITH SUBSTANTIAL MARGIN

REQUIREMENT	MUADEE BUS
15 HZ	21 HZ
28 KSI (ALUM)	+1.8
40 KSI (GR/E)	+ HIGH
< 80 KG (GOAL)	77 KG

PBW: 2/14/94
CHART 2

INTERDEPARTMENTAL COMMUNICATION

TXA-6216

To: P. Williams	Orgn: SET	Bldg: 107	Date: 24 Feb 1994
From: P. Gruver	Orgn: 74-11	Bldg: 150	Ext: 32620
L. Tong			
S. Morrison			

Subject: MUADEE Final Report Charts and Thermal Mathematical Model Description

Attached are the charts for the Thermal Control section of the MUADEE final report. The charts have been updated to reflect changes incurred during the most recent thermal analysis and design effort. Accompanying each chart is a facing page which includes a textual description of the chart contents.

The MUADEE spacecraft thermal mathematical model (TMM) was developed in SINDA. The TMM includes 77 diffusion or real nodes, 60 arithmetic nodes and one boundary node. A spacecraft surface geometry model was also developed to calculate surface view factors and orbital heat fluxes to the surfaces. This model is comprised of 104 active surface nodes.

Each of the Science Instruments (SIs) is conductively coupled to a platform which in turn is conductively coupled to the spacecraft bus. The PIP platform is coupled to the Communications bay and utilizes heat from this bay to warm the platform. Likewise, the NAP platform is coupled to the GNP bay and uses heat dissipated from the GNP electronics to warm the NAP platform.

The SIP platform utilizes some of the heat dissipated from the C&DH bay. The residual heat from this bay is radiatively dissipated to space via a radiator.

The TMM includes nodes for the NiCd battery. The battery is housed in a separate equipment bay and is cooled during discharge via radiation to space from a dedicated louver radiator. When the battery is not discharging or dissipating heat, the louver radiator closes.

Lawrence Tong for
Phil Gruver

Lawrence Tong
Larry Tong

S. Morrison
Susan Morrison

Approved:

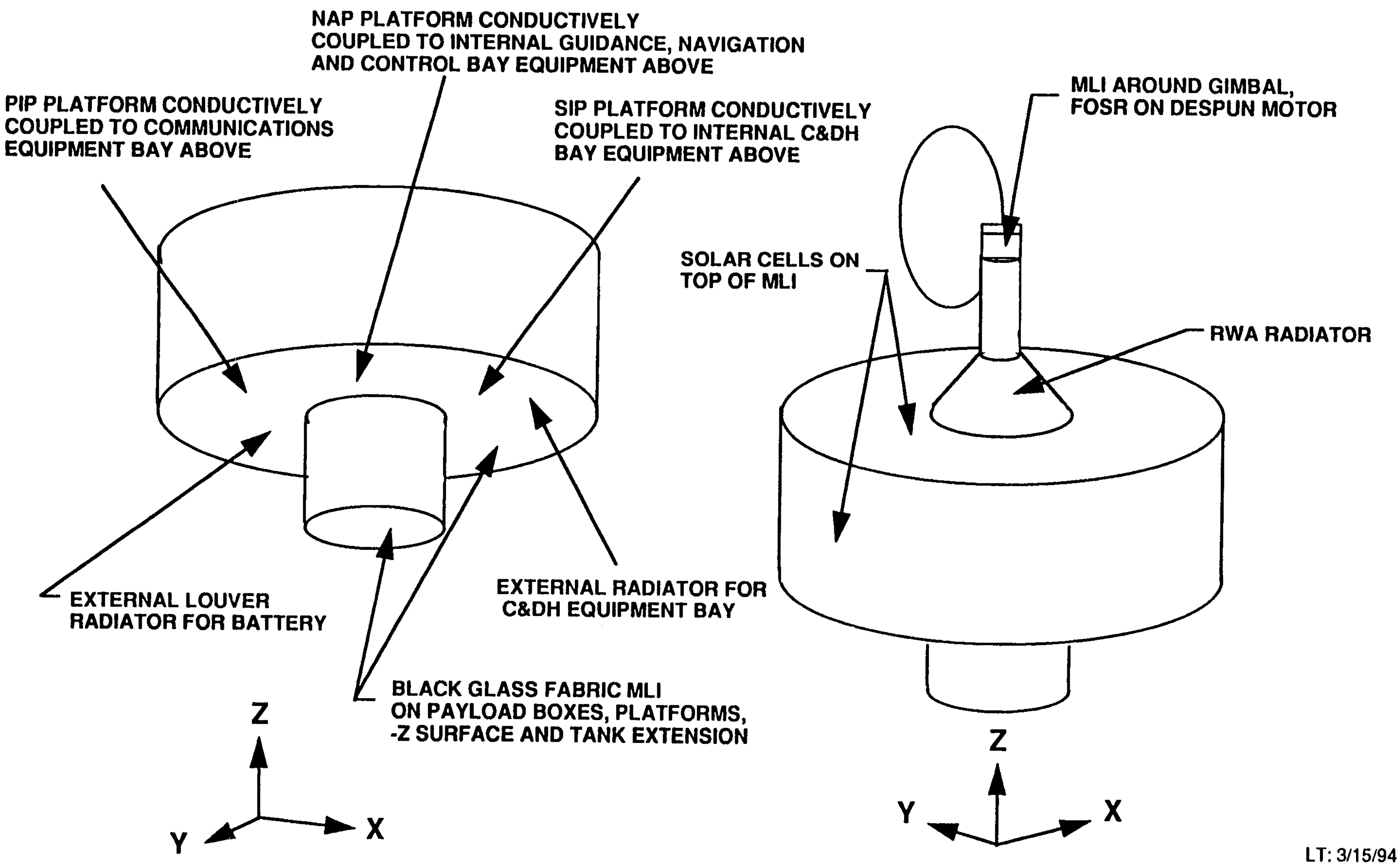
Jim Schirle
Jim Schirle, Group Engineer

TCS ANALYSIS ASSUMPTIONS

ASSUMPTION	JUSTIFICATION
MARS FLIGHT ENVIRONMENT	
NO DIRECT SOLAR FLUX ON -Z SIDE OF VEHICLE	FLIGHT SCENARIO
SOLAR FLUX: 141 TO 209 BTU/HR/SQ-FT	BASED ON MARTIAN ORBIT APOAPSIS AND PERIAPSIS
ALBEDO FACTOR: 0.29	NOMINAL VALUE
PLANETSHINE: 39.5 BTU/HR/SQ-FT	NOMINAL VALUE
BETA ANGLE RANGE: ±90°	BASED ON ONE ORBIT PRECESSION EVERY 60 TO 70 DAYS
CIRCULAR ORBIT OF 200 KM ALTITUDE	NOMINAL ORBIT
VEHICLE RATE OF SPIN: 1°/SEC	TYPICAL PBUS RATE OF SPIN
VEHICLE DESIGN	
SOLAR ARRAY THERMALLY ISOLATED FROM BUS	MINIMIZE EFFECTS OF SOLAR ARRAY TEMP SWING ON BUS
S/C HEAT REJECTION THROUGH -Z SURFACES	NO SUN ON -Z SURFACES
SIs CONDUCTIVELY COUPLED TO EQUIPMENT BAYS	MINIMIZE HEATER POWER BY UTILIZING HEAT FROM EQUIPMENT BAYS
MLI EFFECTIVE EMITTANCE: 0.002 TO 0.02	BASED ON TEST VALIDATION AND FLIGHT EXPERIENCE
HIGH EMITTANCE COATING ON S/C INTERIOR	MINIMIZE THERMAL GRADIENTS WHILE MAXIMIZING HEAT REJECTION
HEATERS ON BATTERY, PROP TANK AND EPS BAY	TIGHT TEMPERATURE TOLERANCES OR WIDE THERMAL EXTREMES

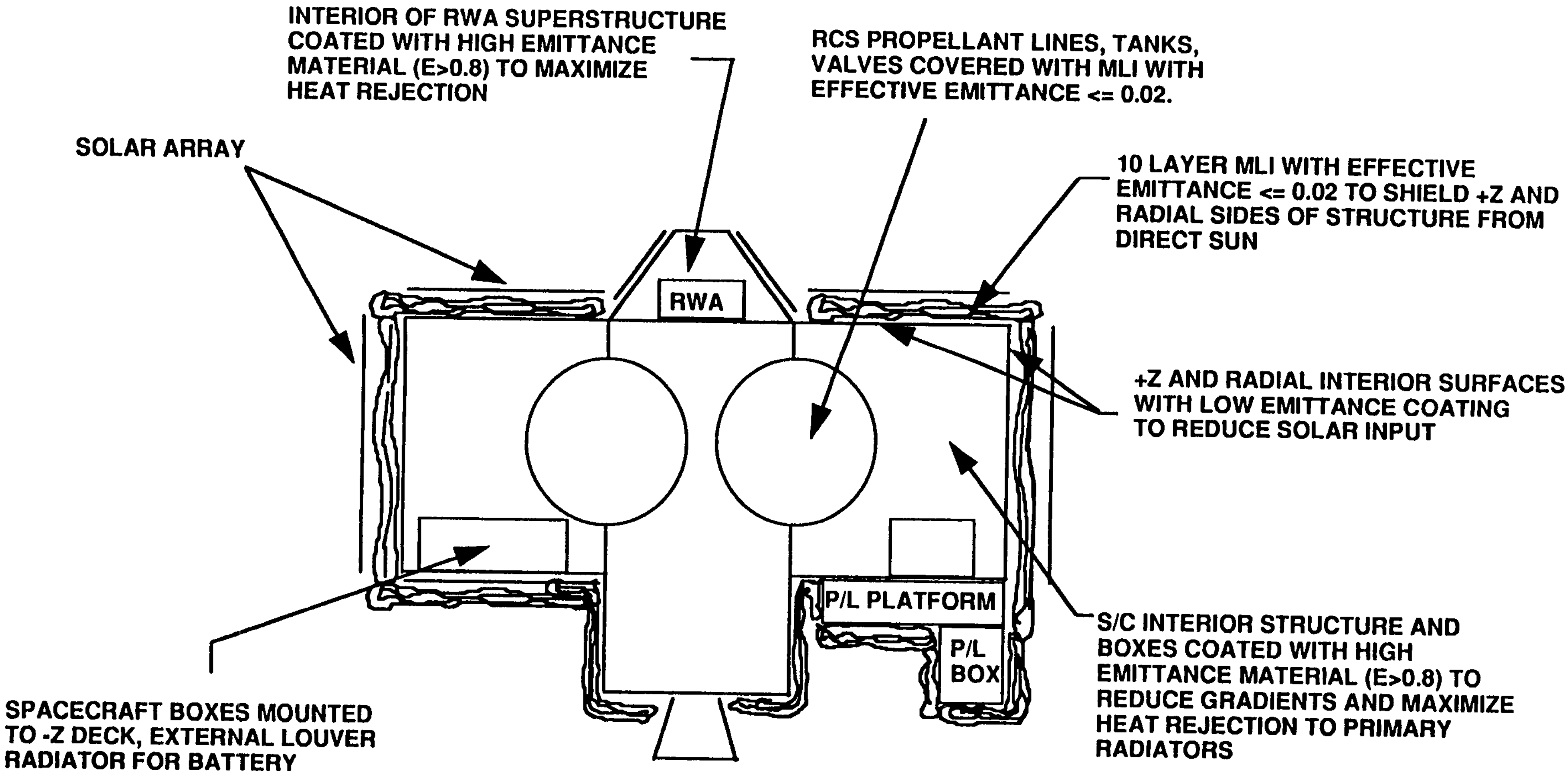
MUADEE

TCS EXTERNAL DESIGN DETAILS



LT: 3/15/94
CHART 5

TCS INTERNAL DESIGN DETAILS



THERMOSTATICALLY CONTROLLED HEATERS ON BATTERY, EPS BAY, COMM BAY AND PROPULSION TANK

SURFACE THERMO-OPTICAL PROPERTIES

	BOL		EOL	
SURFACE FINISH	α	ε	α	ε
ALUMINIZED KAPTON	0.12	0.04	0.16	0.04
BLACK KAPTON	0.8	0.8	0.8	0.8
FOSR	0.12	0.8	0.18	0.8
BLACK MLI	0.92	0.84	0.94	0.84
SOLAR CELLS	0.87	0.86	0.87	0.86
WHITE PAINT	0.5	0.87	0.22	0.87

SJM: 3/15/94

THERMAL MODEL DESCRIPTION

- SINDA MODEL INCLUDES:

  77 DIFFUSION (REAL) NODES
  60 ARITHMETIC NODES
  1 BOUNDARY NODE

- SURFACE GEOMETRY MODEL INCLUDES:

  104 ACTIVE EXTERNAL SURFACE NODES

- THERMAL MODEL OVERVIEW:

  NiCd BATTERY IN ITS OWN BAY WITH DEDICATED LOUVER RADIATOR

  COMMUNICATIONS BAY CONDUCTS TO PIP PLATFORM

  GUIDANCE AND CONTROL BAY CONDUCTS TO NAP PLATFORM

  C&DH BAY CONDUCTS TO SIP PLATFORM
  C&DH BAY HAS DEDICATED RADIATOR

MUADEE

SPACECRAFT THERMAL PREDICTIONS

TEMPERATURES IN DEGREES CELSIUS

HARDWARE	TEMPERATURE REQUIREMENTS		B00 ORBIT			B90 ORBIT			COMMENTS
	SURVIVAL	OPERATIONAL	DATA	PLAYBACK	STANDBY	DATA	PLAYBACK	STANDBY	
PIP PLATFORM	-30/+60	-20/+40	+13	+42	-1	-8	+29	-21	COMM bay moved under PIP platform
NAP PLATFORM	-30/+60	-20/+40	+20	+8	+6	+7	-9	-11	
SIP PLATFORM	-30/+60	-20/+40	+29	+20	+20	+26	+16	+16	
+Z PLATFORM	-30/+60	-20/+40	-17	-17	-17	+56	+56	+56	
VMAG (on boom)	-30/+60	-20/+40	-20	-24	-24	-20	-24	-24	
GUIDANCE AND CONTROL BAY	-24/+62	-24/+62	+21	+9	+6	+7	-8	-11	
C&DH BAY	-24/+62	-24/+62	+29	+20	+20	+26	+16	+16	
COMMUNICATIONS BAY	-24/+62	-24/+62	+13	+42	0	-8	+30	-21	
BATTERY BAY *	+0/+20	+0/+20	+8	+8	+8	+15	+15	+15	Louver added to BATT bay
EPS BAY	-24/+62	-24/+62	-23	-16	-18	-23	-23	-23	Up to 12W heater power reqd during B90 orbit
PROPULSION TANK	+5/+38	+5/+38	+10	+9	+9	+29	+18	+18	Up to 8W heater power reqd for prop tank

NOTE: TEMPERATURE PREDICTIONS DO NOT INCLUDE ANALYSIS UNCERTAINTY MARGIN OR 25% HEATER POWER MARGIN

*ASSUMES A CONSTANT 25W BATTERY DISSIPATION DURING DISCHARGE AND NO DISSIPATION DURING CHARGE

SJM: 3/15/94

INTERDEPARTMENTAL COMMUNICATION

TO: P. WILLIAMS **ORGN:** N-260 **DATE:** 24 Feb. 1994

FROM: C.R.MURDOCK **ORGN:** 74-13 **BLDG:**104 **PHONE:** 25243

SUBJECT: MAUDEE Electrical Power Budget

SUMMARY: In order to ensure the design of the electrical power subsystem (EPS) will meet the load requirements during all mission phases, an electrical power budget must be developed and maintained. The power requirements of each user must be defined, a maturity contingency factor applied, and an appropriate management reserve contingency determined. For the MAUDEE program, all of these steps have been taken. An electrical power budget has been established and the EPS has been designed to provide an average 25% total contingency during mission operations.

This document has been prepared as supporting documentation for the MAUDEE Final Report.

DISCUSSION: Chart 1 of the enclosure, titled "Mission Power Demand", presents a summary of the power requirements of each subsystem and the payload and includes their individual contingency factors. These factors were determined by the responsible design organizations and are based on the maturity of the equipment. Data is also provided for the three key operational modes of the equipment; i.e., science or data taking, playback, and stand-by. As shown on the chart, the average contingency for the total spacecraft load varies from 12% to 15% depending on the operational mode. For each mode an additional management reserve contingency is added to bring the total contingency at the bus power level to 25%. The management reserve factor is applied to ensure the power subsystem capability as designed will meet the load power requirements with a margin of assurance.

Charts 2 presents the power demand for each payload instrument during the data taking mode. Charts 3 through 6 present the power demand for each spacecraft subsystem by component for each orbital mode. The subsystem contingency factors are also shown.

CONCLUSION: An electrical power budget was established for the MAUDEE program which provided the basis for the design power requirement for the EPS. A total 25% contingency over the base load will assure the EPS will meet the mission power demand.

C R Murdock

C. R. Murdock

Enclosure: 6 Charts

MUADEE

MISSION POWER DEMAND

SUBSYSTEM	DATA-TAKING POWER (WATTS)	DATA-TAKING CONT. (%)	PLAYBACK POWER (WATTS)	PLAYBACK CONT. (%)	STAND-BY POWER (WATTS)	STAND-BY CONT. (%)
C & T	9	10	37	10	9	10
C & DH	35	14	35	14	35	14
ACS	54	9	26	16	26	16
MECHANISMS	11	10	11	10	11	10
PROPULSION	1	10	1	10	1	10
THERMAL	6	20	6	20	6	20
EPS	10	20	10	20	7	20
ELEC. DISTR.	8	10	7	10	4	10
S/C TOTAL	134	12	133	14	99	15
PAYLOAD	37	10	0	0	0	0
TOTAL LOAD	171	12	133	14	99	15
MGT. RESERVE	22	13	15	11	22	11
TOTAL BUS POWER	193	25	148	25	109	25

CONT. = CONTINGENCY

CRM 2/24/94
CHART 1

MUADEE

PAYLOAD POWER DEMAND

PAYLOAD INSTRUMENT	OPERATIONAL ORBIT		
	DATA-TAKING	PLAYBACK	STANDBY
NEUTRAL ATMOSPHERE PACKAGE (NAP)	20.1 W		
NEUTRAL / ION MASS SPECTROMETER (NMS)	12.0 W		
FABRY - PEROT INTERFEROMETER (FPI)	8.1 W		
SCANNING IMAGING PACKAGE (SIP)	6.5 W		
ULTRAVIOLET SPECTROMETER (UV S)	3.0 W		
IMAGING PHOTOMETER (IP)	1.5 W		
PLASMA INSTRUMENT PACKAGE (PIP)	10.0 W		
ION DRIFT METER / RETARDING POTENTIAL ANALYZER (IDM)	8.0 W		
LANGMUIR PROBE / EXTREME ULTRAVIOLET SENSOR (LP)	6.0 W		
VECTOR MAGNETOMETER	3.0 W		
TOTAL	36.6 W	0 W	0 W

NOTE: SHADING SHOWS VALUES USED FOR TOTAL.
* ESTIMATED

CRM 2/24/94
CHART 2

C & T POWER DEMAND

COMPONENT	CON-TINGENCY INCLUDED	OPERATIONAL ORBIT POWER DEMAND		
		DATA-TAKING (Watts)	PLAYBACK (Watts)	STAND-BY (Watts)
X-BAND TRANSPONDER	10%	9	18	9
RF AMPLIFIER	10%	0	19	0
TOTAL AVERAGE WATTS:		9	37	9
COMPOSITE CONTINGENCY:		10%	10%	10%

CRM 2/24/94
CHART 3

MUADEE

C & DH POWER DEMAND

COMPONENT	CONTINGENCY INCLUDED	PM-1 DATA-TAKING (Watts)	CHEM PLAYBACK (Watts)	AM-2 STAND-BY (Watts)
TELEMETRY UNIT	15%	15	15	15
COMMAND UNIT	15%	5	5	5
FLIGHT COMPUTER (HONEYWELL)	5%	7	7	7
SOLID STATE DATA RECORDER	20%	6	6	6
OVEN-CONTROLLED OSCILLATOR	10%	2	2	2
TOTAL AVERAGE WATTS:		35	35	35
COMPOSITE CONTINGENCY:		14%	14%	14%

CRM 2/24/94
CHART 4

MUADEE

ACS POWER DEMAND

COMPONENT	CONTINGENCY INCLUDED	OPERATIONAL ORBIT POWER DEMAND		
		DATA-TAKING (Watts)	PLAYBACK (Watts)	STAND-BY (Watts)
HORIZON SCANNERS	5%	7	7	7
REACTION WHEEL	5%	28	0	0
RATE GYROS	20%	19	19	19
STAR TRACKERS (FOR 30 MINUTES EACH 24 HOUR TIME PERIOD)	10%	0	0	0
TOTAL AVERAGE WATTS:		54	26	26
COMPOSITE CONTINGENCY:		9%	16%	16%

MUADEE

MISC. S/C COMPONENT POWER DEMAND

COMPONENT	CON-TINGENCY NCLUDED	OPERATIONAL ORBIT POWER DEMAND DATA-TAKING (Watts)	PLAYBACK (Watts)	STAND-BY (Watts)
PROPULSION	10%	1	1	1
THERMAL CONTROL	20%	6	6	6
EPS (IBE & PDE)	20%	10	10	7
HARNESSES & DIODES	10%	8	7	4
DESPUN & GIMBAL MECHANISM	10%	11	11	11
TOTAL AVERAGE WATTS:		36	35	29
COMPOSITE CONTINGENCY		15%	16%	16%

INTERDEPARTMENTAL COMMUNICATION

To: P. Williams **Orgn.** 74-11 **Bldg.** 107 **Fac.**1 **Date:** 2/28/94

From: R.W. King **Orgn.** 74-11 **Bldg.** 580 **Fac.**5 **Ext.:** 2-5775

Subject: MUADEE Propellant Budget

References: "MUADEE Mars Upper Atmosphere Dynamics Energetics, and Evolution Mission", Draft Final Report, February 7, 1994

The propulsion subsystem is defined in Section 6.11 of the referenced report. It consists of two subsystems, the main propulsion subsystem that provides the propulsive forces for trajectory maneuvers and the reaction control subsystem (RCS) that provides propulsive forces for attitude control.

The following table presents the propellant budget required for the main propulsion subsystem. The table lists the operation, the delta velocity required for the operation, the required propellant , and the propellant budgeted. The operations are listed in chronological order. The delta velocity requirements for the main propulsion subsystem are defined in Section 6.3, Trajectory Design, of the referenced report. The propellant budget has been estimated for a vehicle that has a dry mass of 480 Kg. Performance was assumed to be 315 sec for the 460N NTO/Hydrazine orbit adjust thruster. The budgeted propellant includes an 8% contingency over the required propellant and is assumed to be consumed during each operation.

Main Propulsion Subsystem Propellant Budget

OPERATION	Delta-V (M/s)	Propellant Reqd (Kg)	Propellant Budgeted (Kg)
Launch Dispersion Correction	40	10.71	11.57
Mar Orbit Insertion	1419.5	302.50	326.70
Periapsis Corridor Control	30	4.78	5.16
Transfer to Lower Orbit #1	7.9	1.25	1.35
Orbit Change Maneuver #1	3.1	0.49	0.53
Transfer to Lower Orbit #2	3.1	0.49	0.53
Orbit Change Maneuver #2	20.7	3.25	3.51
Orbit Change Maneuver #3	11.9	1.86	2.01
Orbit Change Maneuver #4	11.8	1.84	1.98
TOTAL	1548.00		353.33

The RCS propellant load was estimated to be 31 Kg. The reaction control delta velocity requirements were assumed to be 8% of the total delta velocity requirement or 134.6 M/s. The propellant budget for this requirement was estimated by assuming that the majority of the RCS burns occur while the vehicle is in Mars orbit. The thruster performance was assumed to be 220 sec. No contingency was added to the required propellant load.

INTERDEPARTMENTAL COMMUNICATION

To: P. Williams **Org.:** 6N-02 **Bldg.:** 107 **Date:** 28 Feb. 1994

From: D. H. Utter **Org.:** 74-16 **Bldg.:** 580 **Phone:** x26275

Subject: MUADEE Pointing Error Budget

Summary:

This report summarizes the MUADEE pointing error budget for several mission phases and control modes.

Discussion:

The pointing error budgets were generated for the trajectory control maneuvers, aerobraking, elliptical orbit, and circular orbit phases of the MUADEE mission. Error budgets for the elliptical and circular orbit phases have been generated for two Attitude Control Subsystem (ACS) modes: a spin mode and a 3-axis stabilized mode. Each error budget contains a detailed breakdown of the contributing terms for both pointing knowledge and pointing accuracy, which are two of the four terms of pointing error requirements. The other two terms, pointing stability and pointing jitter, are not evaluated here because of lack of design maturity. Values for these two top level errors are compared against the requirements and a margin is computed.

For these error budgets, pointing accuracy is the top level. The variance of pointing accuracy is the computed as the sum of the variances of pointing knowledge and pointing control, which are assumed to be independent random variables, Normally distributed. Each value in the error budget is given as three standard deviations, or 3 sigma, (where sigma is equal to the square root of variance) in degrees.

A summary of all the different error budgets is given in Figure 1. The mission phase is shown in column 1. The control mode, either spin mode or 3-axis stabilized mode, is shown in column 2. Pointing knowledge totals from lower level budgets are given in column 3. Pointing control error totals from the lower level budgets are given in column 4. The pointing accuracy total, requirement, and margin are given in columns 5, 6, and 7, respectively.

It can be seen from Figure 1 that the only case in which the pointing accuracy requirement is not met is the elliptical orbit, spin mode case. The requirement for pointing accuracy for the aerobraking case has not been established yet.

Figures 2 through 7 contain the detailed lower level error budgets for each mission phase and control mode combination.

Figure 2 gives the detailed lower level error budget for the Trajectory Control Maneuver case. For this case, the vehicle ACS is in the Spin mode, and the primary attitude information is from the star scanners and sun sensors. Gyros are used in addition to keep track of an inertial attitude reference after the last star update. With the assumption that gyro drift was calibrated within the last 24 hours to be less than 0.01 degrees per hour, then the total pointing knowledge error becomes 0.37 degrees, 3 sigma, which is slightly more than the required 0.3 deg. The pointing control errors are taken from a 6 degree-of-freedom simulation analysis, which shows that spin mode steady-state errors are 0.25 degrees and pointing control of disturbances is on the order of 0.05 degrees. These errors are conservatively assumed to be additive instead of RSS'd because they are not known to be independent. With these assumptions the total pointing accuracy is 0.47 degrees, which meets the requirement of 0.5 degrees.

Figure 3 gives the detailed lower level error budget for the Aerobraking case. For this case, the vehicle ACS is in the 3-axis stabilized, tail first mode. Attitude is determined primarily from the horizon sensors in pitch and roll, with yaw attitude determined by a combination of a last inertial fix from the star scanners and updated attitude changes from the gyros, which have drifted up to 24 hours following a calibration. The pointing knowledge total is exactly the same as that required, 0.3 degrees, 3 sigma. The pointing control error is dominated by the response of the vehicle to aerodynamic torques at an altitude of 115 km. Since aero stability data for the tail-first configuration is not yet available, the response is assumed to be like that of Magellan, which experienced up to 14 degrees excursions during aerobraking at Venus, with a similar aerodynamically stable vehicle and under-powered attitude control system from the reaction control thrusters. The requirement for pointing accuracy has not yet been determined.

Figure 4 gives the detailed lower level error budget for the Elliptical Orbit, Spin mode case. Attitude is determined primarily by horizon crossing indicators and star scanners. Sun sensor data will also be used. The total pointing knowledge is 0.34 degrees, 3 sigma, slightly more than the requirement. Pointing control is dominated by the steady-state spin error of 0.25 degrees added to the error in controlling the yaw aero torque at 130 km. The pointing accuracy total error of 0.62 is slightly larger than the requirement of 0.5 degrees.

Figure 5 gives the detailed lower level error budget for the Elliptical Orbit, 3-Axis stabilized mode. Attitude is determined primarily by horizon static sensors, mounted on stepper motors to cover the wide range of altitude, for pitch and roll, and gyro attitude reference for yaw, with a star scanner inertial fix. The pointing knowledge exactly meets the requirement of 0.3 degrees, 3 sigma. The pointing control error is dominated by the error in controlling the yaw aero torque at 130 km. The total pointing accuracy of 0.40 meets the requirement of 0.5 degrees.

Figure 6 gives the detailed lower level error budget for the Circular Orbit, Spin mode. Attitude is determined as in the Elliptical Orbit, Spin mode. The pointing control errors are given by the addition of the 0.25 degree steady-state spinning error and the error of 0.1 degree in controlling the aero torques at 250 km altitude. Both control errors were determined by simulation analysis. The total pointing accuracy is 0.49 degrees, which barely meets the required 0.5 degrees.

Figure 7 gives the detailed lower level error budget for the Circular Orbit, 3-Axis stabilized mode. Attitude is determined as in the Elliptical Orbit, 3-Axis stabilized mode. Pointing control error is 0.1 degrees as determined by simulation analysis at 250 km altitude. The total pointing accuracy of 0.32 degrees is below the requirement of 0.5 degrees.

Devin Utter
SSD Spacecraft Control Products, O/74-16

MUADEE Error Budget Totals

MISSION PHASE	CONTROL MODE	POINTING KNOWLEDGE (deg, 3-sigma)	POINTING CONTROL (deg, 3-sigma)	POINTING ACCURACY TOTAL (deg, 3-sigma)	POINTING ACCURACY REQUIREMENT (deg, 3-sigma)	POINTING ACCURACY MARGIN (deg, 3-sigma)
Trajectory Control Manuever						
	Spin Mode	0.37	0.30	0.48	0.5	0.02
Aerobraking						
	3-Axis Mode	0.30	14.00	14.00	TBD	unknown
Elliptical Orbit						
2500 by 130 km	Spin Mode	0.34	0.51	0.61	0.5	-0.11
	3-Axis Mode	0.30	0.26	0.40	0.5	0.10
Circular Orbit, 250 km						
	Spin Mode	0.34	0.35	0.49	0.5	0.01
	3-Axis Mode	0.30	0.10	0.32	0.5	0.18

Figure 1.

MUADEE Error Budget TCM

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.37	0.3	-0.07	
	Gyro Reference			0.240				
		drift bias	0.24					24hrs*.01d/h
		mechanical	0.01					LMSC est
		thermal	0.002					temp control
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Sun Sensor			0.253				
		meas process	0.25					Adocole data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
Pointing Accuracy					0.47	0.5	0.03	
	Ptg Knowledge			0.366				
	Ptg Control			0.3				
		spinning error	0.25					spin sim
		control error	0.05					250 km sim

Figure 2.

MUADEE Error Budget Aerobraking

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.30	0.3	0.00	
	Gyro Reference			0.240				
		drift bias	0.24					24hrs*.01d/h
		mechanical	0.01					LMSC est
		thermal	0.002					temp control
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Horizon Sensors			0.150				
	(static sensors)	meas process	0.1					Barnes data
		radiance	0.05					LMSC est.
		mechanical	0.1					stepper motor
Pointing Accuracy					14.00	TBD	unknown	
	Ptg Knowledge			0.303				
	Ptg Control			14				(~ Magellan)
		spinning error	0					
		aero torque error	14					115 km est

Figure 3.

MUADEE Error Budget Spin EII

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.34	0.3	-0.04	
	Sun Sensor			0.253				
		meas process	0.25					Adocole data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Horizon Sensors			0.206				
	(Crossing Indic.)	meas process	0.2					Barnes data
		radiance	0.05					LMSC est
		mechanical	0.01					LMSC est
Pointing Accuracy					0.62	0.5	-0.12	
	Ptg Knowledge			0.344				
	Ptg Control			0.51				
		spinning error	0.25					sim at 250km
		aero torque error	0.26					sim at 130km

Figure 4.

MUADEE Error Budget 3ax EII

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.30	0.3	0.00	
	Gyro Reference			0.240				
		drift bias	0.24					24hrs*.01d/h
		mechanical	0.01					LMSC est
		thermal	0.002					temp control
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Horizon Sensors			0.150				
	(static sensors)	meas process	0.1					Barnes data
		radiance	0.05					LMSC est.
		mechanical	0.1					stepper motor
Pointing Accuracy					0.40	0.5	0.10	
	Ptg Knowledge			0.303				
	Ptg Control			0.26				
		spinning error	0					
		aero torque error	0.26					sim at 130km

Figure 5.

MUADEE Error Budget Spin Circ

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.34	0.3	-0.04	
	Sun Sensor			0.253				
		meas process	0.25					Adocole data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Horizon Sensors			0.206				
	(Crossing Indic.)	meas process	0.2					Barnes data
		radiance	0.05					LMSC est
		mechanical	0.01					LMSC est
Pointing Accuracy					0.49	0.5	0.01	
	Ptg Knowledge			0.344				
	Ptg Control			0.35				
		spinning error	0.25					sim at 250km
		aero torque error	0.1					sim at 250km

Figure 6.

MUADEE Error Budget 3ax Circ

TOP LEVEL	COMPONENTS	SUBCOMPONENTS	SUBCOMP ERROR (deg 3 sig)	COMPON ERROR (deg 3 sig)	TOP LEVEL ERROR (deg 3 sig)	REQT (deg 3 sig)	MARGIN (deg 3 sig)	COMMENTS
Pointing Knowledge					0.30	0.3	0.00	
	Gyro Reference			0.240				
		drift bias	0.24					24hrs*.01d/h
		mechanical	0.01					LMSC est
		thermal	0.002					temp control
	Star Scanner			0.108				
		meas process	0.1					Ball data
		mechanical	0.01					LMSC est
		thermal	0.04					LMSC est
	Horizon Sensors			0.150				
	(static sensors)	meas process	0.1					Barnes data
		radiance	0.05					LMSC est.
		mechanical	0.1					stepper motor
Pointing Accuracy					0.32	0.5	0.18	
	Ptg Knowledge			0.303				
	Ptg Control			0.1				
		spinning error	0					
		aero torque error	0.1					sim at 250km

Figure 7.

LOCKHEED MISSILES & SPACE COMPANY, INC.

ENGINEERING MEMORANDUM

TITLE: Trajectory Design for MUADEE Mission	EM NO.
	REF:
PREPARED BY: R. J. Lynch	DATE: 3/10/94
CHECKED BY:	APPROVAL:

INTRODUCTION

This EM documents the status of the trajectory design for the MUADEE mission. The trajectory design includes departure from Earth, interplanetary transit, capture at Mars and an orbit phase at Mars. The orbit phase at Mars includes several different orbits to achieve the science objectives. This EM covers work completed to date, planned approach for in-depth analysis, and a summary of analysis tools. Preliminary results are included as appendices.

Status

Analysis Tools

For trade offs involving simple, well defined relationships, iterative analysis code is easily generated. These outputs tend to be straightforward plots of families of curves. For investigating orbital behavior at Mars, an existing ephemeris propagator (ICAP) has been adapted to use Mars modelling constants. For the more complex modelling of the interplanetary trajectory, a specialized tool (INTRPLNT) will developed by Lockheed specifically for MUADEE . This tool provides the capability to evaluate the effect of launch dates, mission duration's, mid course corrections, and departure and arrival conditions.

REQUIREMENTS

The following science requirements were provided by the University of Michigan (Ref. 5).

1) Obtain height profiles of the thermosphere and the ionosphere via in-situ measurements.
2) Obtain global ultra-violet images of the planet from very high altitudes.
3) Obtain limb scanning height profiles of the atmospheric emission lines in both the ultra-violet and visible spectrum.
4) Obtain altitude profiles of the thermosphere and ionosphere measured over a nearly full range of latitudes twice in 6 months.

5) Latitude excursions of periapsis shall occur at nearly fixed local times separated by 12 hours.

Requirements 1,2, and 3 are satisfied by the deep diving phase (section 5.2) of the mission. Requirements 4 and 5 are satisfied by the global survey (section 5.3) phase of the mission.

The following operational requirements were derived by Lockheed.

1) Trajectory design must accommodate launch dates in 1998, 2001 and 2003.

This first requirement allows the spacecraft to be designed to accommodate slips in schedule and other unforeseen events that might delay the launch of MUADEE to a date other than 1998. The trajectory design for 2001 is the ΔV design condition (section 4.8)

2) U.S. launch vehicles only.

This requirement is applied to prevent complications with the involvement of other countries. In fact this requirement can be removed at any time if politics warrant.

3) Maximum time of flight from Earth to Mars of 1 year.
4) On orbit lifetime of 2 years.

These requirement are needed to keep operations and ground support costs to a minimum and to reduce spacecraft design needs for TCM's, orbit maintenance, corrosion, heating, solar panel efficiency, battery condition, etc.

5) Contamination of Mars or it's moons by spacecraft components shall be avoided.

This requirement basically says "Give a Hoot, Don't Pollute"

6) Aerobraking without a heat shield shall be utilized to minimize ΔV.

This requirement says minimize the propellant mass and don't add any additional mass in the form of a heat shield. Development of a heat shield can be extremely costly in dollars and in mass.

7) Trajectory correction maneuvers (TCM) shall be ground commanded.
8) The 1st TCM shall occur no earlier than 2 weeks after launch.

These two requirements ensure that the spacecraft is under the full authority of the ground controllers before any thrusting maneuvers. This is basically a spacecraft health and safety concern.

DELTA II 7925 PERFORMANCE

The performance of the Delta II 7925 for interplanetary trajectories is given in Figure 1 below. The performance is given in kg as a function of C_3. C_3 is the square of the hyperbolic excess velocity. Hyperbolic excess velocity is the velocity over above that required to escape from the Earth's sphere of influence. Thus at a C_3 of 0.0 km^2/s^2 the spacecraft is in orbit about the Sun in the same orbit as the Earth. To change the spacecraft's orbit excess velocity must be applied. The data is for a 28.5 degree inclination orbit. Typical Mars missions require a launch C_3 of between 10 - 12 km^2/s^2. For a C_3 of 12 km^2/s^2 the Delta II 7925 payload capability is 1,025 kg.

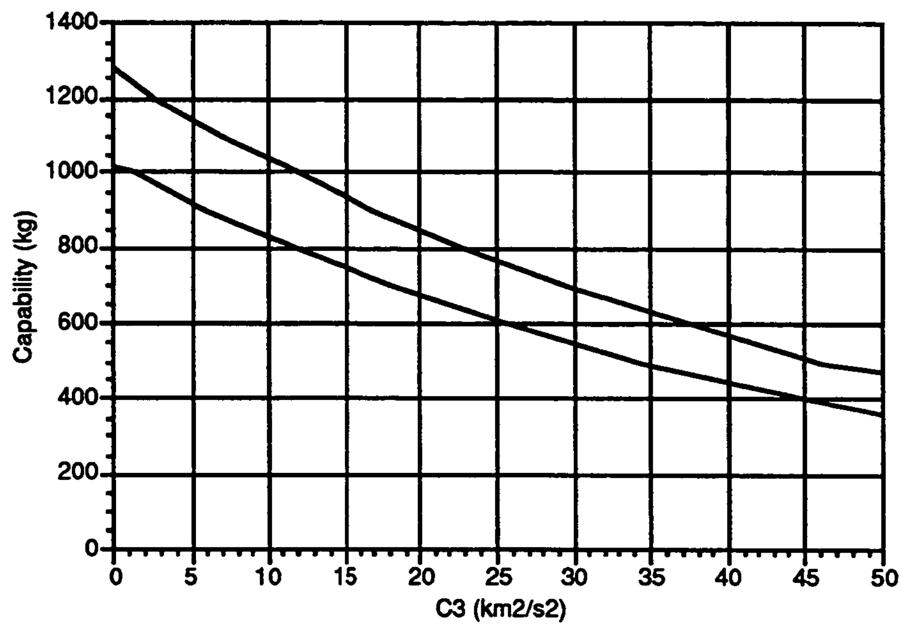


Figure 1 Delta II Performance Capability

4.0 INTERPLANETARY TRAJECTORY

OVERVIEW

There are two basic problems that need to be solved in order to determine the interplanetary trajectory. They are Kepler's problem and Lambert's problem. Kepler's problem is also known as the trajectory propagation problem. Given a know initial state on a trajectory find the final state at some later time. The solution to Kepler's problem is unique. Lambert's problem is also known as the trajectory targeting problem. Given a know initial position and a desired final position as well as the time of flight between the two positions, find the departure and arrival velocities. There are an infinite number of solutions to Lambert's problem. To obtain a unique solution the transfer angle and direction must be specified.

For the MUADEE mission the interplanetary trajectory is defined to be a direct transfer. A direct transfer does not employ planetary gravity-assist flybys. Direct transfers can be either ballistic or non-ballistic. Direct non-ballistic transfers have velocity changes (ΔVs) between the launch from Earth and Arrival at Mars. Direct ballistic transfers correspond to coasting trajectories. The trajectory for the MUADEE spacecraft was determined by assuming a ballistic transfer. However, provisions are included for trajectory correction maneuvers which employ that the actual transfer is non-ballistic. Direct transfers are classified according to their total transfer angle Θ, between launch from Earth and arrival at Mars.

Transfer Type	Transfer Angle (deg)
Type I	0< Θ >180
Type II	180< Θ > 360
Type III	360< Θ > 540
Type IV	540< Θ >720

Table 4.1.1 Direct Transfer Types

C3 REQUIREMENTS

Figure 4.2.1 below illustrates two important quantities in interplanetary trajectory design C_3 at launch and arrival. C_3 is the square of the hyperbolic excess velocity. The hyperbolic excess velocity is the difference between the departure/arrival planets heliocentric (sun centered coordinate frame) velocity and the heliocentric velocity of the spacecraft at departure/arrival. C3 is plotted as contour plots for each of the launch years being considered in figures 4.2.2 thru 4.2.4.

TBS

Figure 4.2.1 Definition of Launch and Arrival Energy (C_3)

TBS

Figure 4.2.2 C_3 as a Function of Launch and Arrival Date for 1998

TBS

Figure 4.2.3 C_3 as a Function of Launch and Arrival Date for 2001

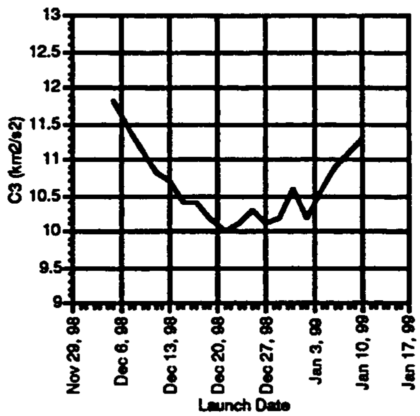
TBS

Figure 4.2.4 C_3 as a Function of Launch and Arrival Date for 2003

LAUNCH OPPORTUNITIES

Based on the C_3 plots of section 4.2 the launch opportunities can be defined. The launch opportunities are determined by finding the optimal launch period for each opportunity. The optimal launch period will minimize the required C_3 thus maximizing the payload weight, and minimize the ΔV required for Mars Orbit Injection (MOI) thus minimizing propellant weight. Another important consideration is the launch geometry of the outgoing hyperbolic asymptote. This is discussed in section 4.5 below.

Figures 4.3.1 thru 4.3.3 show the required C_3 and MOI ΔV vs launch date for each of the launch opportunities.



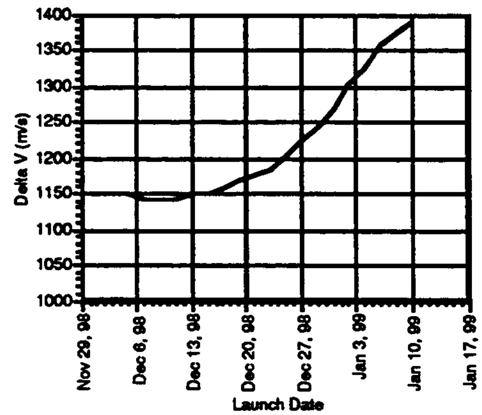


Figure 4.3.1 C_3 and MOI ΔV as a Function of Launch Date for 1998

From figure 4.3.1 the optimal launch period for 1998 is found to be 12/6/98 to 12/28/98 with the optimal launch date on 12/13/98.

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Figure 4.3.2 C_3 and MOI ΔV as a Function of Launch Date for 2001

From figure 4.3.2 the optimal launch period for 2001 is found to be TBD to TBD with the optimal launch date on TBD.

TBS

Figure 4.3.3 C_3 and MOI ΔV as a Function of Launch Date for 2003

From figure 4.3.3 the optimal launch period for 2001 is found to be TBD to TBD with the optimal launch date on TBD.

From the above figures the launch opportunities are defined and are shown in figure 4.3.4 below. The VEGA (Venus Earth Gravity Assist) opportunity shown in 2004 was obtained by a literature search and analysis to confirm it's existence has not been completed. However, due to the trajectory design groundrules the analysis of the VEGA opportunity has not been a high priority, it is shown here for completeness. The launch opportunities shown are for ballistic transfers and do not consider limitations based on the launch geometry. Launch geometry is discussed in the next section.

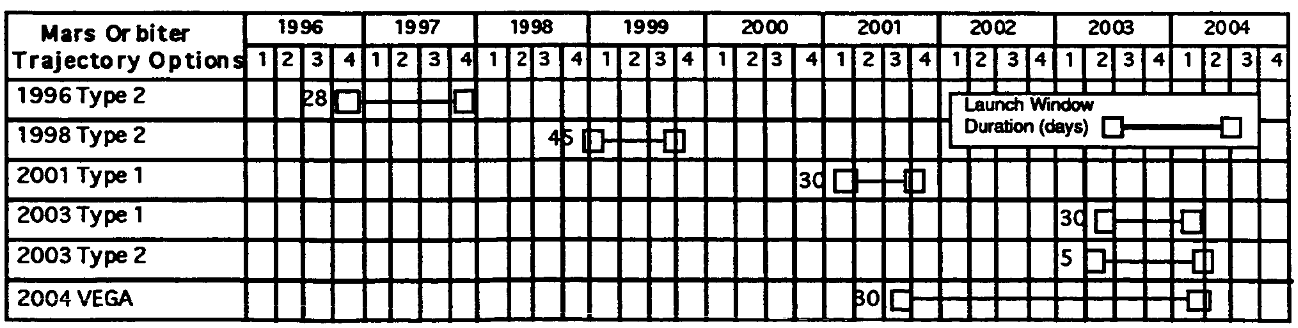

Figure 4.3.4 Launch Opportunities

LAUNCH GEOMETRY

The primary problem in departure trajectory design is to match the required hyperbolic V-infinity vector (the vector in the direction of the velocity on the hyperbola at an infinite distance) with a specified launch site location on Earth. The launch site is defined by its geocentric latitude ϕ_l=28.5, and its geographic longitude λ_l=279. Range safety considerations prohibit over flight of populated areas by the ascending launch vehicle. The sector of allowed azimuth firing directions for the Delta II launch vehicle from Kennedy space center complex 17 is shown in figure 4.4.1 below. For the range of azimuths between 65 - 100 deg the typical orbit inclinations that the booster can achieve are between 15 - 57 deg. For maximum performance the inclination should be equal to the pad latitude.

Furthermore the inclination of the orbit should be greater than the declination of the V-infinity vector. It should be noted that as the declination of the V-infinity vector increases the contribution to the spacecraft's escape velocity due to the Earth's rotation becomes smaller. The penalty for a V-infinity vector declination that is greater than the orbit inclination is 31.5 kg/deg. The launch geometry is shown in figure 4.4.2 below. Another important angle is the right ascension of the V-infinity vector. The correct right ascension and declination are achieved by the use of a launch window. The launch window is typically 30 - 60 minutes long and represents the extremes of the launch vehicles capability to obtain the correct right ascension and declination. The variation of the declination of the V-infinity vector over the launch period for the various launch opportunities are given in figures 4.4.3 thru 4.4.5.

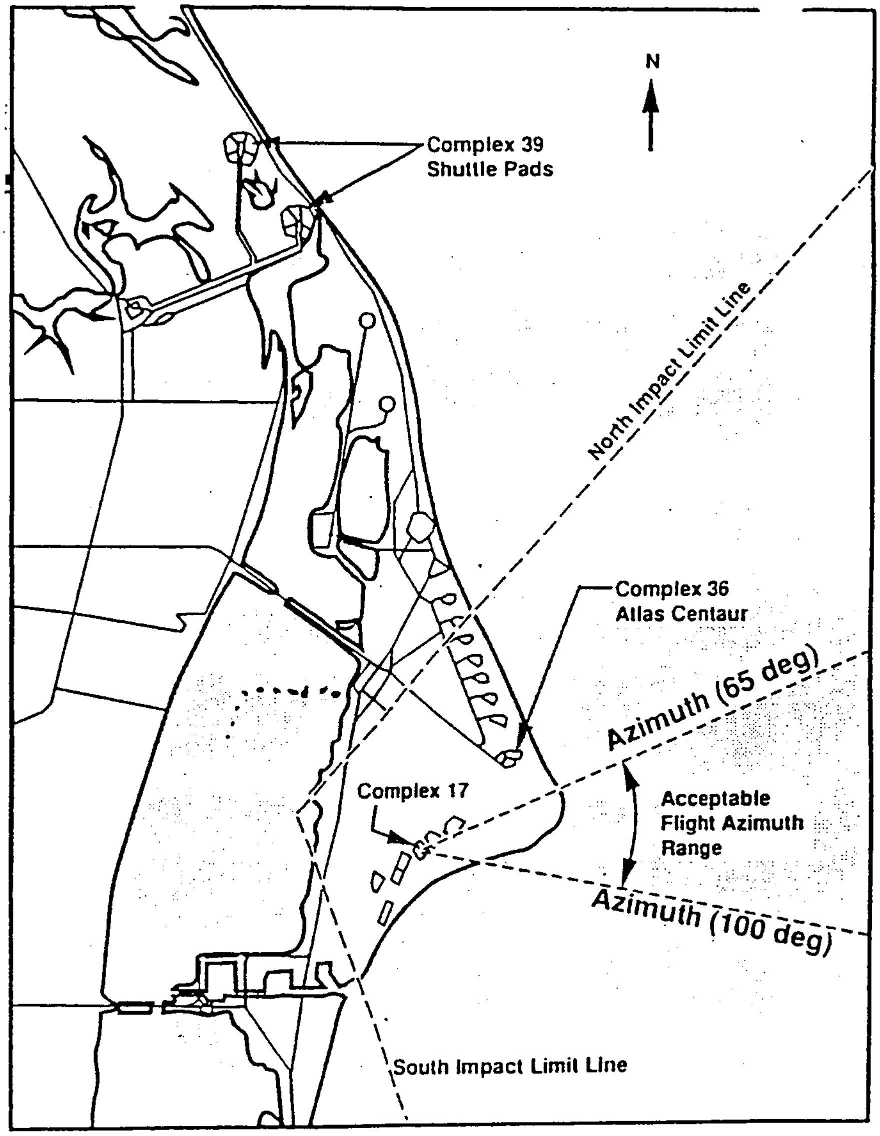

Figure 4.4.1 Launch Azimuth Range Restrictions

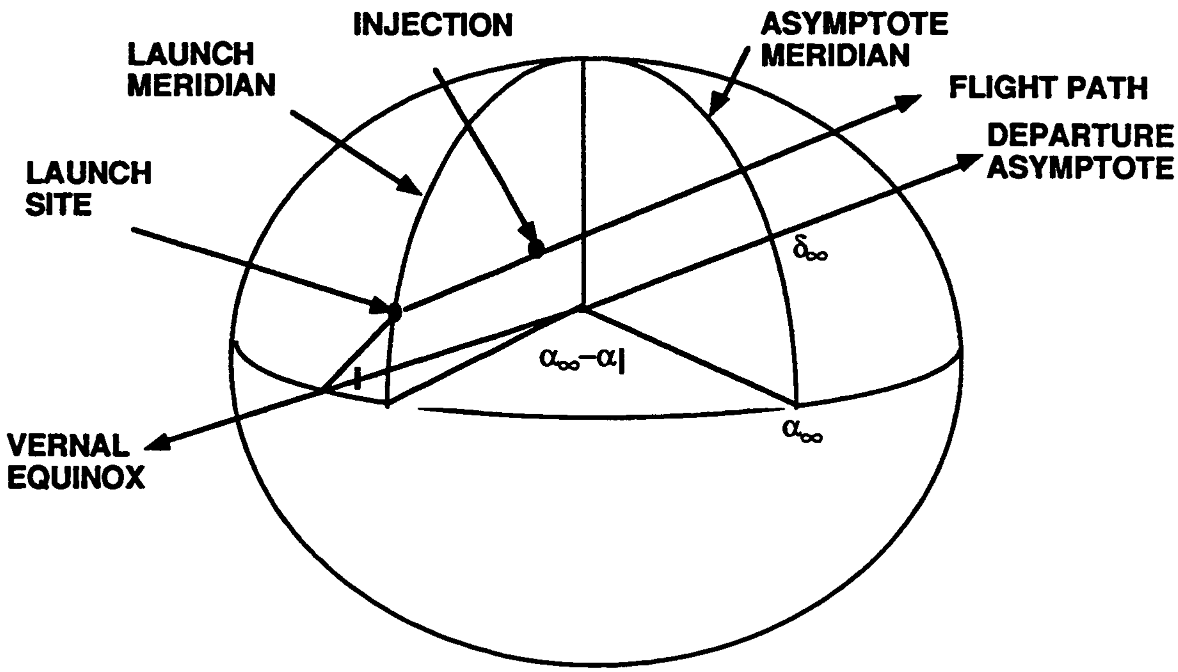

Figure 4.4.2 Launch Geometry

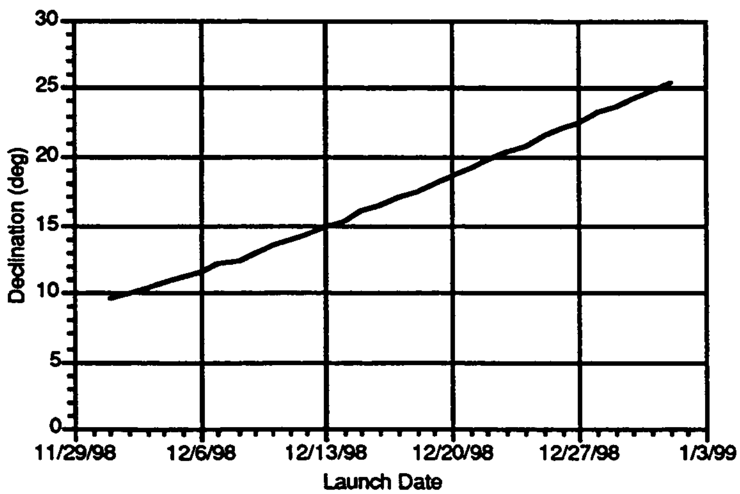

Figure 4.4.3 Declination Variation for 1998

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Figure 4.4.4 Declination Variation for 2001

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Figure 4.4.5 Declination Variation for 2003

ARRIVAL GEOMETRY-TBS

MARS ORBIT INSERTION-TBS

INTERPLANETARY REFERENCE TRAJECTORY

The interplanetary trajectory orbit parameters are provided in Table 4.7.1. The parameters are for the optimum launch date of 12/13/98 and are referenced to the heliocentric coordinate frame.

Parameter	Earth at Launch	Mars at Arrival	Transfer
Date	12/13/98	9/29/99	N/A
Semi-Major Axis (km)	149,597,870	0.0934	178,721,892
Inclination (deg)	0.0167	1.85	0.1858
Argument of Perihelion (deg)	102.9	336.1	2.77
Long. of Ascnd. Node (deg)	0.0	49.6	22.5
True Anomaly (deg)	337.5	196.5	81.28

Table 4.7.1 Reference Trajectory Parameters for 1998

INTERPLANETARY ΔV BUDGET

Description	ΔV (m/s)
Launch Dispersions	120.0
Trajectory Correction Maneuvers (TCM)	30.0

MARS ORBIT DESIGN

CAPTURE ORBIT- TBS

DEEP DIVING

The requirements to obtain height profiles of the thermosphere and ionosphere via in-situ measurements, global ultraviolet images of the planet from very high altitudes, and limb-scanning height profiles of the atmospheric emission lines in the ultra-violet and visible spectrum are satisfied by the deep diving phase of the mission. The deep diving phase of the mission is a result of the 1 day capture orbit which has an apoapsis altitude of 33,100 km and a periapsis altitude of 250 km and the desire to use aerobraking to lower this orbit to the 250 km altitude circular orbits required for the diurnal phase of the mission (see section 5.4). Due to the atmospheric structure of Mars the periapsis altitude must be lowered to an altitude of approximately 110 km before significant drag forces occur. This highly eccentric orbit allows limb scanning and ultraviolet images of the planet to be taken from very high altitudes and also allows for the in-situ measurements required to obtain height profiles of the thermosphere and ionosphere.

After aerobraking has lowered apoapsis to 2,500 km the periapsis will be raised to 130 km. This elliptical orbit will be inclined at 95.5 degrees to obtain altitude profiles of the thermosphere and the ionosphere over a full range of latitudes. The near polar inclination of the orbit causes the orbit plane to precess slowly with respect to the fictitious mean sun location. This causes both latitude excursions of periapsis to occur at nearly fixed local times that are separated by 12 hours.

GLOBAL SURVEY- TBS

ECLIPSES- TBS

AEROBRAKING- TBS

ORBIT HISTORY- TBS

MISSION PROFILE- TBS

TOTAL ΔV BUDGET

Description	ΔV (m/s)
Mars Orbit Insertion (MOI)	1,419.5
Transfer to Lower Orbit #1 (TLO-1)	7.9
Orbit Change Maneuver#1 (OCM-1)	3.1
Transfer to Lower Orbit #2 (TLO-2)	3.1
OCM-2	20.7
OCM-3	11.9
OCM-4	11.8
Periapsis Corridor Control	30.0
Launch Dispersions	40.0
Trajectory Correction Maneuvers (TCM)	30.0
TOTAL	**1,578.0**

PLANETARY PROTECTION- TBS

TRAJECTORY SENSITIVITY

Interplanetary trajectory design and understanding of the issues has now matured to the point where a more detailed sensitivity analysis can be undertaken. The flight to Mars can be divided into three regimes: Earth departure, interplanetary transit, and Mars arrival. Variances in the vehicle's trajectory in each stage, will affect the trajectory accuracy in the next. A number of points along the trajectory are significant for evaluating trajectory dispersions, either as sources of error or as target points with associated allowable dispersions. These are the interfaces between the three trajectory regimes, as well as the mission events listed above. In addition, there is an "aim point" related to the Mars approach which is very useful in the trajectory design.

The accuracy requirements during Mars orbit are established in response to the science objectives of the mission. For optimizing these objectives, the various orbital parameters can be specified and tolerances associated. Detailed examination of the orbital motion over time will determine the expected variances in the orbit and the design costs of compensating for them.

The accuracy requirements throughout transit are ultimately driven by the mission requirements at Mars. For this reason, the sensitivity analysis will address the stages of transit in reverse order. The required accuracy of the capture orbit at Mars will define the allowable dispersions throughout the Mars approach phase and at the transition from interplanetary transit. Similarly, the targeted conclusion of the interplanetary stage will define the tolerance throughout transit and at insertion. Again working backward, tolerances can be established for the Earth departure stage of the trajectory.

In each regime, the sensitivity analysis will identify parameters for examination. Some will be used as measurements to determine a range of acceptable cases; other parameters are of interest because they are controllable and can be used to determine the cost of mission accuracy.

Once the relevant parameters are selected, trade studies can be pursued which evaluate the dependencies among them. In this way, such things as the cost of meeting an allowed tolerance or the penalty for exceeding it can be quantified. For example, for the desired capture periapsis altitude of 250 km, the approach trajectory must be aimed at an altitude of 2200 km. This sensitivity analysis will be used to determine the cost of correcting various dispersions from the 2200 km altitude and the allowable variation in the aim point to achieve a given tolerance in the periapsis altitude. Such relationships will be examined for each stage of the trajectory.

The method for determining the tolerances and dispersions can be as sophisticated as necessary. In most cases, sufficient useful results will be produced analytically by varying the inputs to a known relationship and calculating the outputs. This type of approach is easily implemented in an iterative algorithm. As greater refinement is required, other methods of error analysis can be applied.

The results of these sensitivity studies will provide input to the design process. Alternative mission objectives or requirements can be evaluated for their impact in related areas and trade studies will be supported for design decisions

PLANETARY CONSTANTS - TBS
COORDINATE FRAMES- TBS

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